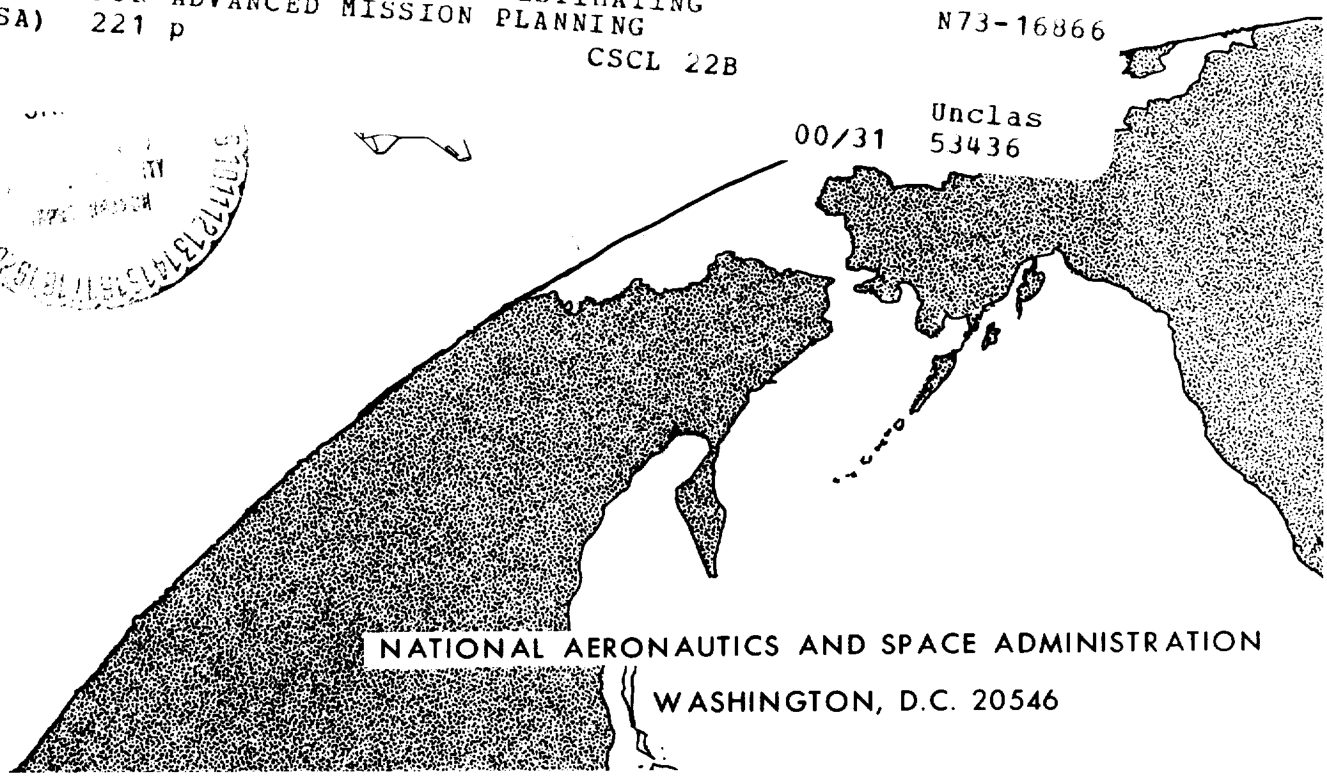
LAUNCH VEHICLE ESTIMATING FACTORS FOR ADVANCE MISSION PLANNING

NHB 7100.5B

1973 EDITION



NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D.C. 20546

PREFACE

This revised launch vehicle reference document presents a systematic arrangement of information useful in the advance planning of space missions. Requirements for a variety of solar system and Earth orbital missions are described. Performance capabilities and characteristics are presented for current expendable launch vehicles, for near-future expendable launch vehicles, the space shuttle, and for possible future launch vehicles and stages incorporating solar-electric, nuclear-electric, and nuclear-thermal propulsion systems.

This 1973 edition presents new information on the drift of Sun synchronous orbits. Planetary launch opportunity data have been extended to the end of the century. Revised data are presented for the space shuttle and other launch vehicles. Recent changes of the nomenclature for Delta launch vehicles are incorporated.

The information contained herein is sufficiently accurate for advance planning by NASA and other government agencies. However, the data should not be used for detailed mission planning without the concurrence of the Director of Launch Vehicle and Propulsion Programs of the Office of Space Science. This document is reviewed and updated annually, but the actual or target performance of elements of the space transportation system are subject to change at any time. Any questions concerning this document or the applicability of the data may be directed to Mr. J. E. McGolrick, Mr. B. C. Lam, Mr. J. A. Salmanson, or Mr. J. W. Haughey at NASA Headquarters, Office of the Director of Launch Vehicle and Propulsion Programs (Code SV), Telephone 202-755-3726.

NHB 7100.5A is hereby superseded.

Joseph E. McGolrick
Advanced Programs and Technology
Program Manager

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BLEED INDEX

CHAPTER

BLEED INDEX, CONT.

CHAPTER 1: INTRODUCTION

TABLE OF CONTENTS

CHAPTER 1: INTRODUCTION

100 PURPOSE

This document presents a concise and systematic arrangement of information useful for the preparation of advance mission plans. The information can be used for establishing first estimates of launch-vehicle requirements for a broad variety of space-flight opportunities.

101 SCOPE

1. Advanced planning of potential space missions requires comprehensive knowledge of both mission characteristic data and launch-vehicle performance data. This document presents mission characteristic data for two basic categories of space-flight opportunities. These are: solar-system missions and Earth-orbital missions. Performance data for a broad selection of current expendable launch vehicles are presented. In addition, estimated performance data are included for a variety of near-term and future launch vehicles including expendable chemical launch vehicles, the space shuttle, solar-electric propulsion systems and nuclear-thermal and nuclear-electric propulsion systems.

2. The information presented in this document is considered to be sufficiently accurate for advance planning. In no instance should these data be used for detailed mission planning without concurrence of the Director of Launch Vehicle and Propulsion Programs of the Office of Space Science (OSS). Questions should be referred to the persons listed in the Preface.

102 ORGANIZATION AND PROCEDURES

1. The chapters in this document are organized so that the advance planning of a mission, having been defined in terms of orbital specification or space destination and payload characteristics, may normally proceed through the following steps:

   a. Determination of basic mission velocity requirements (Chapter 2 and Chapter 3)

b. Determination of launch-site effects and incremental correction of velocity requirements as a function of orbit inclination or launch azimuth at each site (Chapter 4)

c. Determination of the total characteristic velocity (defined in Appendix A) required to accomplish the mission, through combination of the results of Steps a and b

d. Determination of the launch vehicle(s) that can deliver the prescribed payload at the required characteristic velocity from a specified launch site (Chapter 5 for conventional launch vehicles, Chapter 7 for the Space Shuttle, and Chapter 10 for proposed nuclear-thermal vehicles).

This general procedure is appropriate for most kinds of missions; however, because of various launch-vehicle constraints, Earth-orbital missions in particular require special consideration. Chapter 6 includes data on Earth-orbital performance capabilities for selected expendable launch vehicles. Chapter 7 includes performance data for a space shuttle both with and without transfer stages.

Federal law and NASA policy require planners to define and consider potential safety and environmental hazards which might arise from their proposed activities at all stages of the planning activity, and to consider alternative methods of accomplishing the desired objective which might reduce or eliminate these hazards. Applicable information concerning small and medium-sized launch vehicles is available in the Environmental Statement for the National Aeronautics and Space Administration, Office of Space Science, Launch Vehicle and Propulsion Programs, and from the individuals named in the Preface.

Similar information concerning the Space Shuttle can be obtained from NASA Code MH.

2. Velocity packages may be used to increase the characteristic velocity of some launch vehicles with small payloads. Performance data and physical characteristics for selected solid-propellant velocity packages are presented in Chapter 8.

3. Chapter 9 gives performance data for solar-electric propulsion systems. Chapter 10 presents data for nuclear-thermal and nuclear-electric propulsion systems.

4. Chapter 11 provides information useful for estimating kick-stage requirements for applications such as planetary orbiter retro-propulsion systems and apogee kick stages.

5. Chapter 12 presents line drawings of launch-vehicle-shroud configurations. These data are useful for the determination of nominal payload-physical-dimension constraints.

6. Appendix A is a brief glossary of the terms used in this document. All references in this document are listed in Appendix B.

103 UNITS

NASA and other agencies and organizations currently have adopted the International System (SI) of Units. This system is used in the present document. Conversion factors to English units are noted on figures and charts as appropriate. In keeping with NASA policy, the users of this document are urged to familiarize themselves with and to use the International System of Units. Information on the International System of Units may be found, for example, in Reference 1 (Appendix B).

CHAPTER 2: SOLAR SYSTEM MISSION FACTORS

TABLE OF CONTENTS

Figures-cont.

Figures-cont. | Page

CHAPTER 2: SOLAR SYSTEM MISSION FACTORS

200 INTRODUCTION

1. Chapter 2 presents data for the preliminary estimation of launch-energy requirements for a variety of solar-system or Earth-escape missions. It should be emphasized that these data are intended only as first approximations. Appendix A should be consulted for definitions of terms. For more detailed information, the reader is referred to information available in the references cited throughout this chapter. The references are listed in Appendix B.

2. The performance required to obtain a specific unpowered spacecraft trajectory is most succinctly characterized by the summation of all the discrete and impulsive changes in velocity that must be imparted to the spacecraft. The characteristic velocity obtained in this manner depends only upon the mission itself and the assumed sequence of impulses employed to inject the spacecraft into its final ballistic trajectory.

3. The information in this chapter may be used to obtain the characteristic velocity (V_C) requirements for a variety of missions. By definition, the characteristic velocity for a 185 km (100 n. mi.) circular orbit is equal to the actual orbital speed referred to Earth-centered nonrotating coordinates (7.797 km/sec or 25,581 ft/sec). The characteristic velocity for any other mission is obtained by summing all additional velocity increments required to perform the mission to the orbital speed for the assumed 185 km circular parking orbit.

4. For most solar-system or Earth-escape missions, the characteristic velocity requirements obtained in this chapter may be compared directly to launch capabilities shown in Chapters 5, 7, and 10 provided that the launch is eastward from Cape Kennedy. For launches other than eastward from Cape Kennedy, an appropriate penalty must be obtained from Chapter 4 and added to the basic mission requirements.

201 CAUTIONARY NOTE

Mission-requirements data presented in this chapter are based on specific trajectories. In using these data to determine launch-vehicle requirements, it should be borne in mind that operations involving specific parking orbits, plane changes, and orbit circularization may require staging, coasting, or multiple burning of a stage beyond the capability of a particular launch vehicle. Questions concerning these matters should be referred to persons listed in the Preface.

202 ECLIPTIC-PLANE MISSIONS

1. The characteristic velocity requirements for missions to regions lying in the ecliptic plane are shown in Figures 2-1 through 2-5.

2. Figure 2-1 presents the characteristic velocity data as a function of trip time and distance from the Sun for direct flights to regions farther out from the Sun than Earth (outer ecliptic region). The planets and the asteroid Ceres are represented by bands of distance from the Sun. These bands are intended to show the limits of radial distances caused by the eccentricity of the planet orbits. Table 2-1 provides a summary of important characteristics of the planets and their orbits based on References 2 and 22 (Appendix B).

3. As seen in Figure 2-1, trajectories to regions far from the Sun, with reasonable flight times, require very high launch velocities. These high velocity requirements can be alleviated somewhat by employing Jupiter gravity-assisted trajectories as shown in Figure 2-2. The use of a close encounter with a planet to modify the heliocentric trajectory of a spacecraft has been studied by many investigators, and Jupiter has the most dramatic potential as a swingby target for automated vehicles. Figure 2-2 may be compared with Figure 2-1 to see the rather substantial savings in either launch velocity or trip time that may be obtained using the Jupiter-assisted trajectories to the outer regions of the ecliptic plane.

4. For missions to regions closer to the Sun than Earth (inner ecliptic region), minimum energy direct flights require less than 6 months, so only the minimum velocity requirements are shown in Figure 2-3. Again, the bands of distance from the Sun representing Venus and Mercury are shown for general information only.

5. Gravity-assisted trajectories may be used to advantage under some circumstances for regions close to the Sun. Venus

TABLE 2-1. PLANETARY AND LUNAR CHARACTERISTICS[a]

	Mercury	Venus	Earth	Luna	Mars	Ceres	Jupiter	Saturn	Uranus	Neptune	Pluto
Semi-major Axis, a. u.[d]	.387098	.723332	1.000000	384,400 km[b]	1.523692	2.7675	5.202803	9.538843	19.182110	30.057416	39.373641
Eccentricity	.2056203	.0068064	.0167385	.05490	.0933405	.0759	.0483865	.0548488	.0478624	.0085516	.2488033
Perihelion, a. u.	.307503	.718408	.983262	356,400 km[b]	1.381469	2.5574	4.951058	9.015648	18.264008	29.800377	29.577349
Aphelion, a. u.	.466694	.728255	1.016739	406,700 km[b]	1.665914	2.9776	5.454549	10.062038	20.100212	30.314455	49.169933
Inclination, degrees											
Orbit to Ecliptic	7.003638	3.394023	0.000000	5.1425	1.850070	10.607	1.306528	2.490850	.772939	1.775675	17.169866
Equatorial to Orbit	-- (c)	-- (c)	23.449722	1.52018	25.2000	-- (c)	3.11500	26.74500	97.5900	29.0000	-- (c)
Mean Orbital Velocity(EMOS)[e]	1.607271	1.175794	1.000000		.810124	.60007	.438411	.323782	.228324	.182399	.159367
Period (Earth years)	.240842	.615186	1.000000	0.0748022	1.880813	4.602	11.867413	29.460733	84.012627	164.788714	247.063373
Radius, km	2435	6052	6378	1738	3393	386	71372	60401	23535	22324	7016
Mass (Earth=1)[f]	.054972	.805097	1.000000	0.0123001	.106141	.000215	314.018416	93.992913	14.343655	17.076843	.181512
Gravitational Parameters[g], km^3/sec^2	2.218243+4	3.248724+5	4.035194+5[h]	4.90278+3	4.283006+4	8.61+2	1.267125+8	3.792796+7	5.787941+6	6.890837+6	7.324365+4
Sphere of Influence, Radius, a. u.	.0007534	.0041196	.0062112	65,934 km	.0038586	.0006	.3222507	.3646768	.3457282	.5808836	.0928316
Period of Rotation, Earth days	58.67	242.6	1.0	27.738	1.02596	-- (c)	.41069	.42639	.45069	.5833	6.39
Escape Velocity at Surface, km/sec	4.2684	10.361	11.248	2.372	5.1352	.667	59.588	35.438	22.178	24.846	4.569
No. of Satellites	0	0	1	0	2	0	12	9	5	3	0

(a) Planetary data adapted from Reference 2 (Appendix B); Luna and Ceres data adapted from Reference 22 (Appendix B)

(b) Distance from Earth

(c) Uncertain

(d) 1 a. u. = 149,600,000 km

(e) EMOS = 29,782 m/sec

(f) Mass of Earth = 5.977×10^{24} kg

(g) The + n following each number indicates a power of 10

(h) Value is for Earth + Luna, Earth alone is 3.986012+5 km^3/sec^2

Conversion Factors:
km/sec x 3.28 = 1000 ft/sec
km x 0.540 = n. mi.

gravity-assisted trajectories to Mercury and to the vicinity of 0.2 a.u. have been shown to be superior to direct flight in some cases (see References 3 and 4 in Appendix B). Correspondingly, Jupiter swingbys can be very effective in reducing the large launch velocity required for very close Solar probes. Trajectory data for close Solar probes employing Jupiter swingby are shown in Figure 2-4. The Jupiter swingby mode, however, involves flight times that are significantly greater than the flight times for direct flights.

6. In the event that a heliocentric orbit is desired that will range through distances from the Sun that lie on both sides of Earth's orbit, Figure 2-5 shows the characteristic velocity required to achieve various combinations of perihelion and aphelion distances. This figure can be used, for example, to determine the characteristic velocity requirements for heliocentric orbits that traverse solar system space on both sides of the Earth's orbit.

203 OUT-OF-ECLIPTIC MISSIONS

1. Data for direct flights to regions out of the ecliptic plane are presented in Figure 2-6 that show the required launch characteristic velocity for reaching a given point defined by the celestial latitude and radial distance from the Sun. These are optimum values in the sense that the characteristic velocity has been minimized at all points. The corresponding flight times are also shown.

2. Since any out-of-ecliptic launch requires directing the hyperbolic excess velocity vector away from the ecliptic plane, launch azimuth constraints, which are discussed in Chapter 4, may become significant. The optimum trajectories that require launches outside the nominal azimuth limits from the ETR lie in the shaded region of Figure 2-6. Probes to points within the shaded region probably would have to be launched from the WTR to achieve the high declination angles required. In this case, a launch-site velocity penalty would have to be added to the basic velocity requirement obtained from Figure 2-6. For launches from the ETR within the nominal launch azimuth limits, the velocity penalty incurred by launching in a noneasterly direction is always less than 119 m/sec. Further information is contained in Reference 5 (Appendix B).

3. It is apparent from Figure 2-6 that direct launches to points at high celestial latitudes may require prohibitively large launch velocities. Jupiter swingbys have been shown to be useful in

reducing the launch velocities required for out-of-ecliptic probes. Figure 2-7 shows a plot of the altitude above the ecliptic plane versus distance from the Sun in the ecliptic plane for fixed values of characteristic velocity. These data are based on optimized trajectories in which the magnitude of the characteristic velocity is minimized for each point and the direction of flight leaving Earth and the encounter conditions at Jupiter are unconstrained.

204 PLANETARY FLYBY AND ORBITER MISSIONS

1. For missions to the planets, the relative positions of the Earth and the target planet in their respective orbits must lie within certain angular limits for reasonable launch velocities. Consequently, the synodic period of revolution of the target planet with respect to Earth is of paramount importance in establishing launch opportunities. If the orbits of Earth and the target planet were perfectly circular and coplanar, the launch-energy requirements would be identical at each launch opportunity as established by the synodic period. However, because the planet orbits are neither circular nor coplanar, significant differences can occur in the minimum launch energies required for different launch opportunities. Because of such differences, Figures 2-8 through 2-15 present characteristic velocity data corresponding to particular launch opportunities.

2. Figure 2-8 shows launch characteristic velocity requirements for direct flights to Mercury. Since Earth-Mercury geometry is repetitious, the pertinent launch data shown have an approximate 4750-day cycle. This period is only slightly more than 13 Earth years so that launch opportunities, characteristic velocities, and trip times for direct trajectories in 1990, for example, will be essentially the same as in 1977.

3. For Mercury, three (and sometimes four) launch opportunities occur each year. Figure 2-8 shows only data for the two opportunities each year which are of primary interest. Those annual opportunities having the minimum launch characteristic velocities are indicated by the solid bars in Figure 2-8. For these opportunities, the approach velocities relative to Mercury are large (14 to 19 km/sec). Another opportunity, always different from the preceding one, occurs each year for which the approach velocity is minimum. These opportunities, indicated by the open bars in Figure 2-8, are most appropriate for Mercury orbiter missions.

4. No consideration is given in Figure 2-8 to particular opportunity widths because of the rapidly changing trajectory parameters which are characteristic of Mercury opportunities. A typical velocity increment above the indicated characteristic velocities for a 15-day opportunity would be 122 m/sec. Trade-offs between mission requirements and opportunity widths must be made on an individual mission basis in which it is necessary to consider many more constraining parameters, such as those related to departure and arrival geometry, than can be discussed here.

5. Swingby trajectories employing a gravity assist from Venus have been shown to be beneficial during some years for both flyby and orbiter missions to Mercury. Figure 2-9 shows launch velocity requirements for Venus-swingby missions to Mercury between 1974 and 1987. The bars indicate the launch characteristic velocities for unpowered swingby opportunities which give the lowest approach velocities at Mercury. The absolute minimum launch velocities are generally only slightly less than the values shown. Characteristic velocity increments for 20-day opportunities are small and generally on the order of one percent of the launch characteristic velocity. For more detailed information on the Venus swingby missions, consult References 6, 7, and 8 (Appendix B).

6. Figure 2-10 shows characteristic launch-velocity requirements for Mars and Venus missions. In each case, the velocity increment required for a 30-day launch opportunity and the minimum launch velocity requirement are indicated. For some opportunities, Type I trajectories have the minimum launch-velocity requirements; for others, Type II trajectories have the minimum requirements. In cases where there is a significant trade-off between launch velocity and flight time, data are shown for both Type I and Type II trajectories.

7. Because of excessive flight times, absolute minimum energy launches to Jupiter and the outer planets are not attractive. Therefore, the data in Figures 2-11, 2-12, and 2-13 depict characteristic launch-velocity requirements for particular flight times to Jupiter, Saturn, and Uranus, respectively, along with the velocity increments to permit 30-day launch opportunities.

8. For Neptune, the yearly changes in the launch-velocity requirements are negligible, so the velocity requirements are shown in Figure 2-14 as a function of trip time only.

9. The yearly changes in velocity requirements become significant for probes to Pluto because of the large eccentricity and inclination of Pluto's orbit. Characteristic velocity data for two flight times are shown in Figure 2-15. Only one opportunity per year exists. Figures 2-11 through 2-15 are based upon data presented in Reference 9 (Appendix B). That report should be consulted for further information regarding direct Jupiter, Saturn, Neptune, and Pluto missions.

10. For missions to the planets beyond Jupiter, the use of a Jupiter gravity-assisted trajectory may be advantageous in reducing either trip time or characteristic launch velocity (see References 10, 11, 12, and 13 in Appendix B). Data in Figures 2-16a, b, c, and d show launch-velocity requirements for particular flight times to Saturn, Uranus, Neptune, and Pluto via Jupiter swingby. Similar data for missions to Uranus via Saturn and for Neptune via Uranus are shown in Figures 2-17a and b. The velocity increments for 20-day opportunities are shown because they are significantly less than the increments for 30-day opportunities.

11. Unusual opportunities for multiple planet swingbys occur in the 1975 to 1981 time period. There will be an opportunity for missions using successive Jupiter, Saturn, and Uranus swingbys to Neptune during each year from 1976 to 1980. The unique set of opportunities to perform this mission recurs only every 179 years. Figure 2-18 shows launch-velocity requirements for particular flight times for this mission. Two other multiple-planet swingby combinations of considerable interest are shown in Figure 2-19. These are a Jupiter-Saturn-Pluto mission and a Jupiter-Uranus-Neptune mission. Further information can be obtained from References 8, 15, 16, 17, and 18 (Appendix B).

12. In general, the Jupiter swingby mode will be beneficial for flyby missions, but the increases in approach velocity at encounter with the target planet caused by the higher energy trajectories may preclude the use of the Jupiter swingby mode for orbiter missions.

13. The frequency of opportunities for Jupiter swingby trajectories to the outer planets depends upon the synodic period of the outer planets relative to Jupiter. The synodic periods of the outer planets relative to Jupiter are as follows: 19.8 Earth years for Saturn, 13.7 Earth years for Uranus, 12.7 Earth years for Neptune, and 12.0 Earth years for Pluto.

205 PLANETARY ORBITER ENERGY REQUIREMENTS

1. For planetary orbiters, the magnitude of the required retro-velocity increment that must be provided by the spacecraft propulsion unit depends upon the approach velocity relative to the target planet, the mass of the planet, and the periapsis and eccentricity of the desired satellite orbit. If the only mission requirement is that of being captured by the target planet, an orbit of high eccentricity but very low periapsis is most economical. The required retro-impulse, which is applied at periapsis, is smaller at lower periapsis altitudes. If the retro-impulse is applied at very high periapsis altitudes, the required impulse, at worst, approaches the magnitude of the initial hyperbolic excess velocity relative to the planet.

2. For circular orbits or orbits of small eccentricity, the required retro-impulse is, of course, larger than that required for capture orbits of higher eccentricity. In some cases, an altitude exists that minimizes the required velocity increment for a circular orbit.

3. Data that permit determination of retro-velocity requirements for all potential target planets except Pluto are given in Figures 2-20 through 2-27. The first six of these figures give the approach velocities associated with the trajectories for which launch-velocity requirements were given in Figures 2-8 and 2-10 through 2-14.

4. Figure 2-26 shows escape velocities for different periapses for the planets and the Earth's Moon. The relation between retro-impulse and approach velocity and escape velocity is shown in Figures 2-27a and 2-27b. The following procedure yields the retro-impulse requirements:

   a. Determine the approach velocity from Figures 2-20 through 2-25.

   b. Select the periapsis and apoapsis radii for the desired final orbit. (It is convenient to choose the orbit such that the ratio of apoapsis to periapsis is one of those plotted in Figure 2-27a or 2-27b. Refer to Table 2-1 for planetary radii.)

   c. Determine the escape velocity at periapsis for the planet being orbited from Figure 2-26.

d. Calculate the ratio of apoapsis to periapsis of the final orbit and the ratio of approach velocity to escape velocity at periapsis.

e. Use Figures 2-27a or 2-27b to determine the ratio of retro-impulse to escape velocity at periapsis, and multiply this ratio by the escape velocity obtained in step c.

An estimate of the retro-propulsion system mass can be made by using these data in conjunction with information in Chapter 11.

206 LUNAR-MISSION REQUIREMENTS

Figure 2-28 presents characteristic-velocity requirements and equivalent hyperbolic excess velocity at the Moon for lunar missions as a function of trip time for the Moon at perigee and at apogee. In general, characteristic velocity requirements and approach velocities will lie between these pairs of curves. Retro-propulsion requirements for orbiter missions can be found using the equivalent hyperbolic excess velocity together with data in Figures 2-26 and 2-27a or 2-27b by following the procedure described in paragraph 205.4. If this procedure is used, orbits should be restricted to those with apolunes less than 22 lunar radii (equivalent to an altitude of about 36,300 km). Additional information may be obtained from References 19, 20, and 21 (Appendix B).

207 CONVERSION CHARTS

Figures 2-29, 2-30a, and 2-30b are included for rapid conversion from the characteristic velocity used in this document to hyperbolic excess velocity, Earth Mean Orbital Speed (EMOS), and the energy parameter C_3.

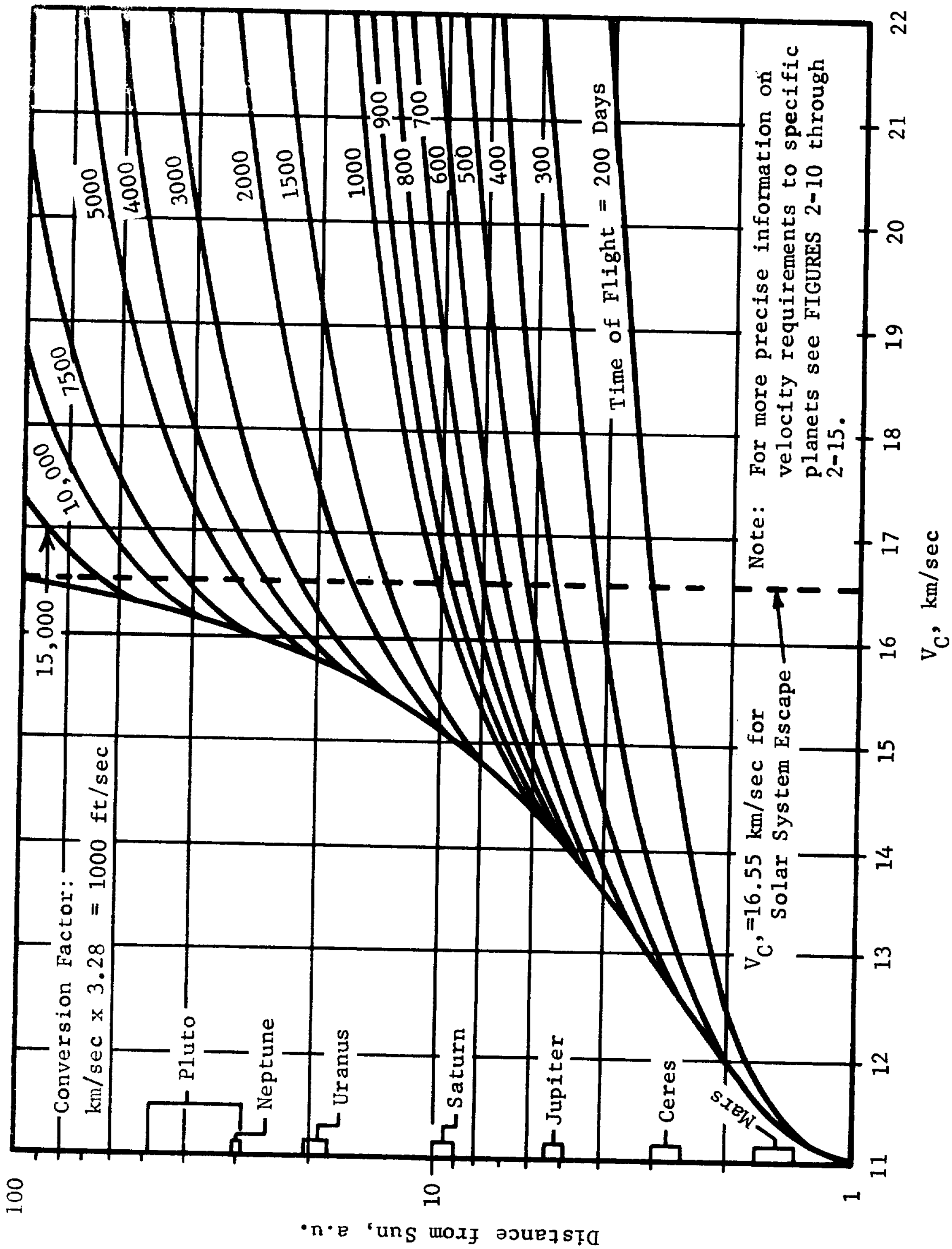


FIGURE 2-1. VELOCITY REQUIRED FOR BALLISTIC PROBES TO OUTER ECLIPTIC REGIONS

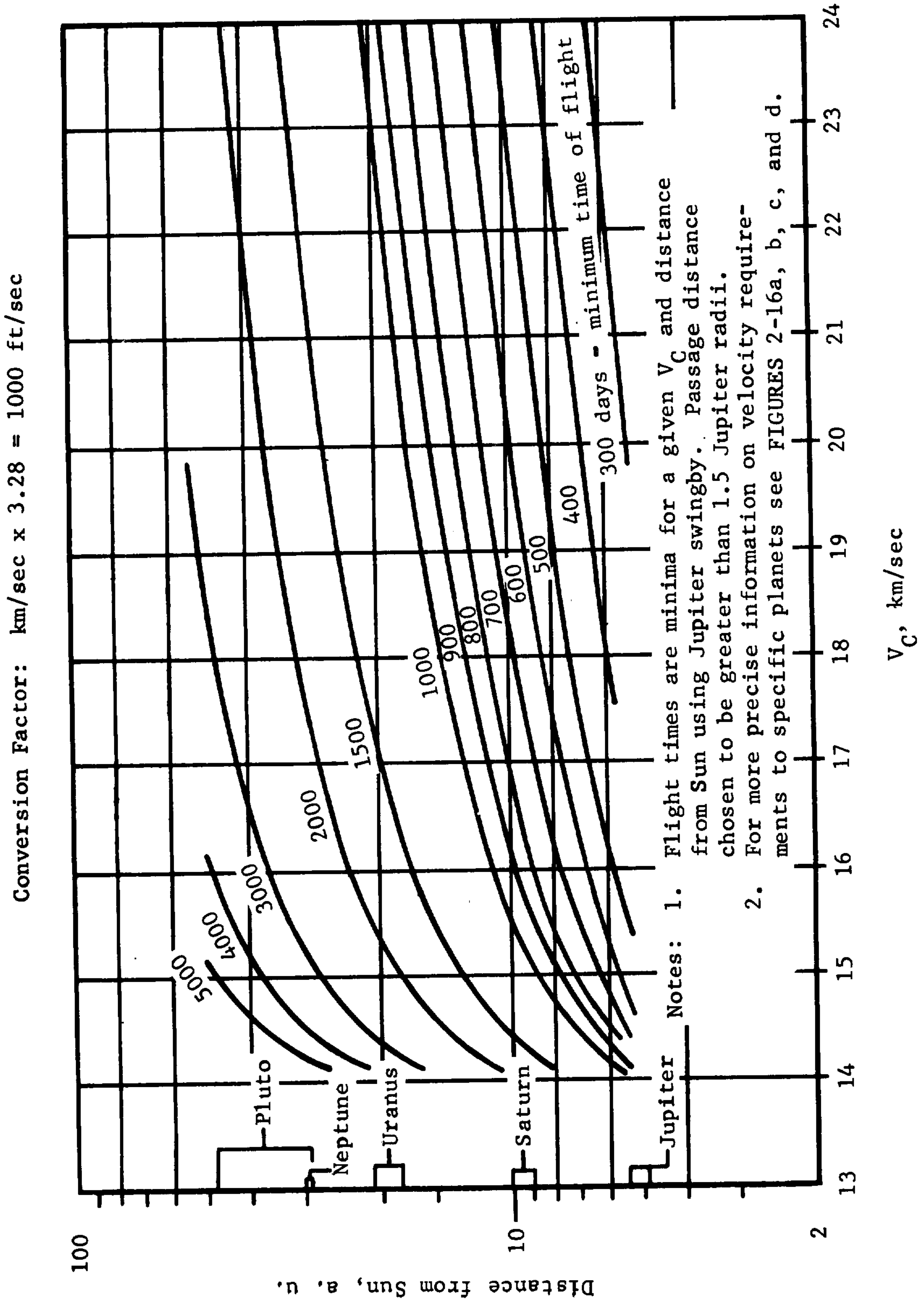


FIGURE 2-2. VELOCITY REQUIRED FOR PROBES TO OUTER ECLIPTIC REGIONS WITH JUPITER SWINGBY

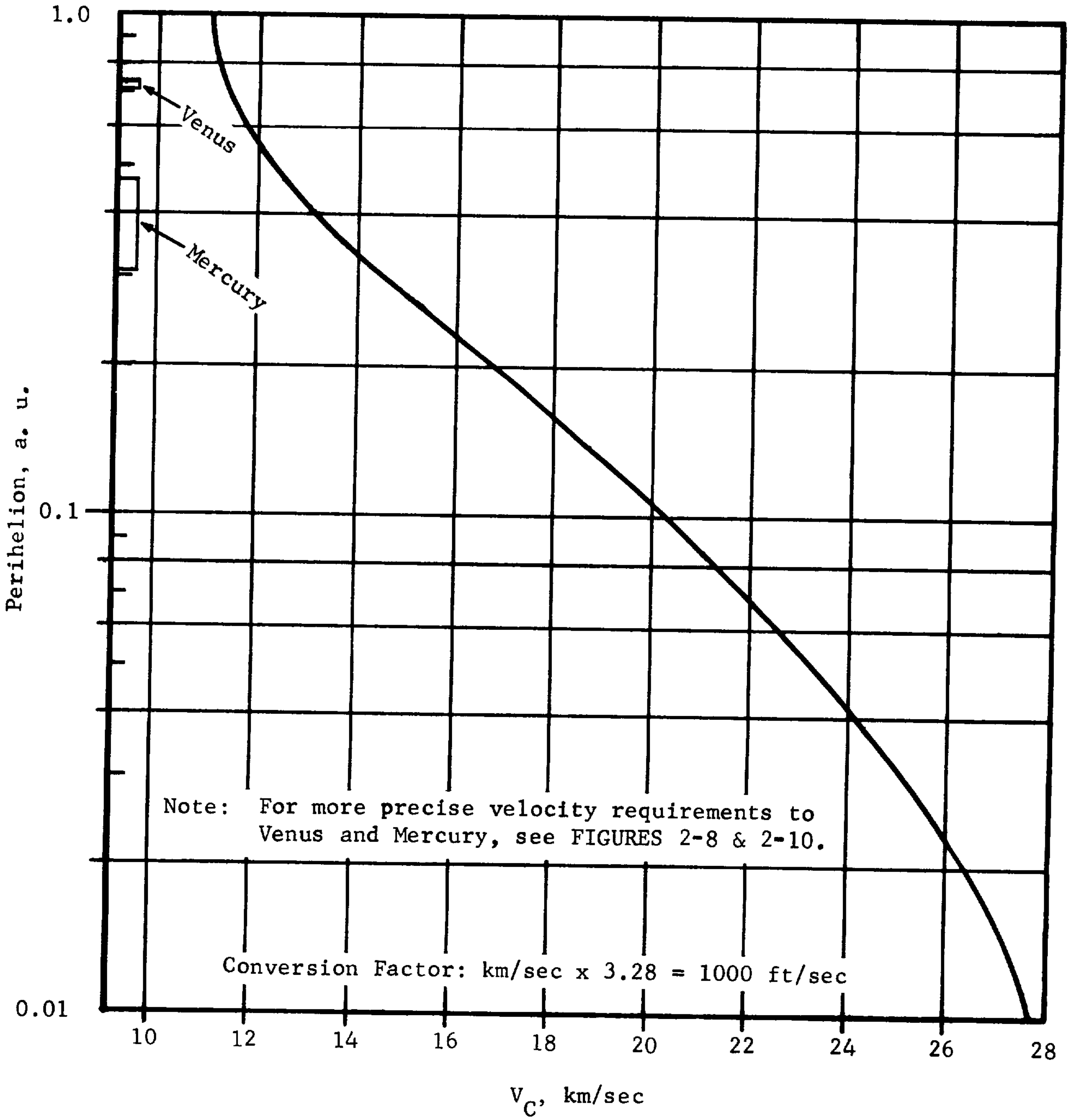


FIGURE 2-3. VELOCITY REQUIRED FOR BALLISTIC SOLAR AND INNER ECLIPTIC PROBES

Conversion Factor: km/sec x 3.28 = 1000 ft/sec

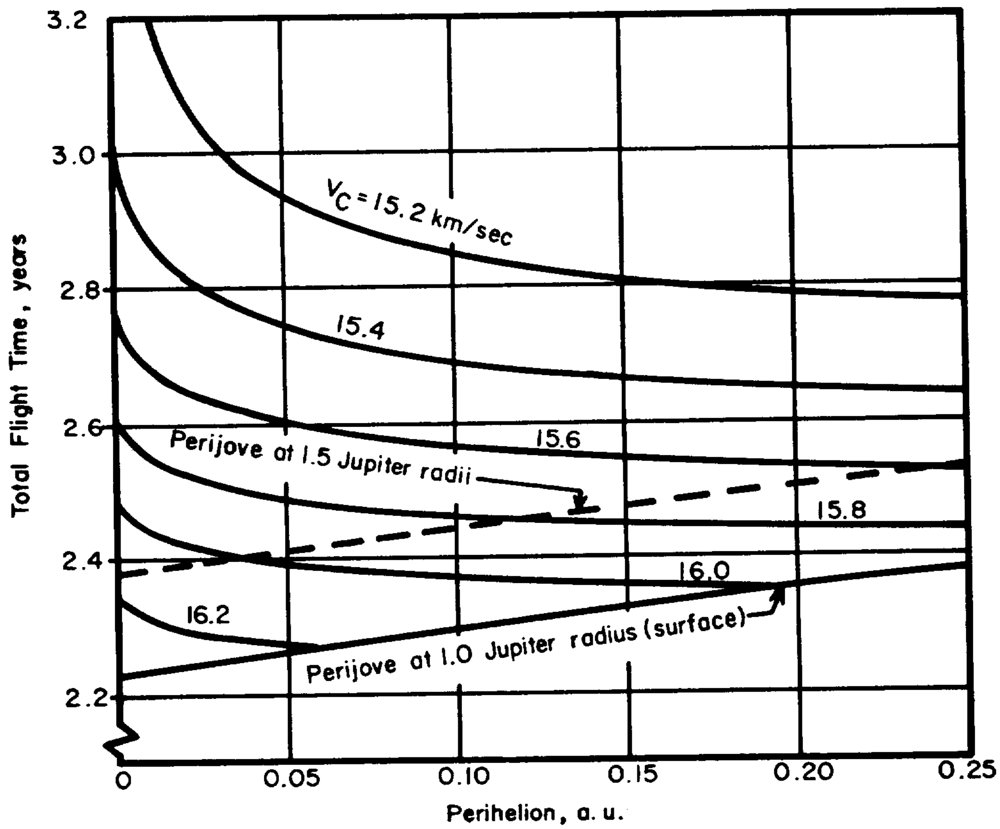


FIGURE 2-4. TRAJECTORY DATA FOR SOLAR PROBES USING JUPITER SWINGBY

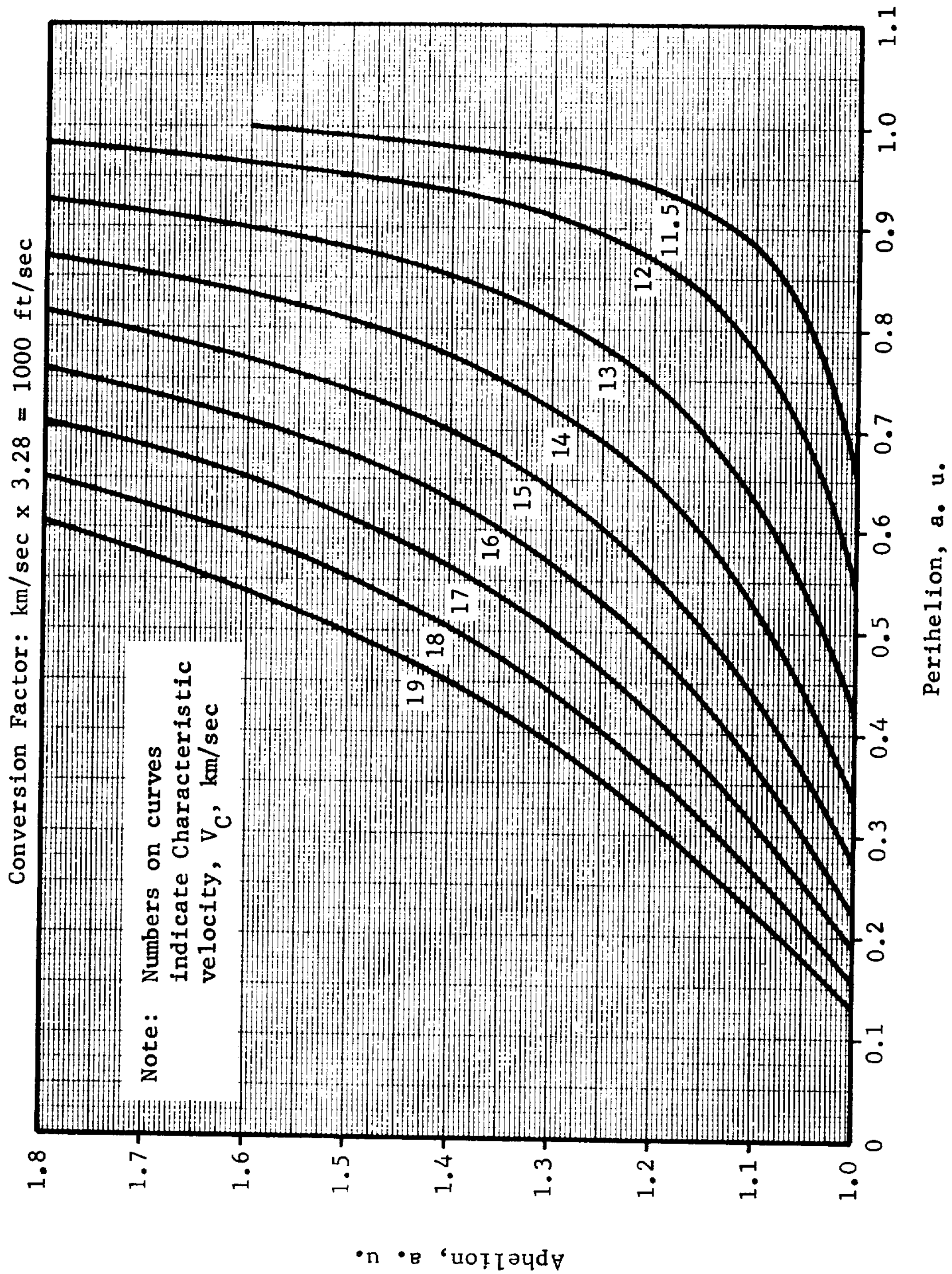


FIGURE 2-5. VELOCITY REQUIREMENTS FOR BALLISTIC PROBES IN THE ECLIPTIC PLANE WITH VARIOUS PERIHELION AND APHELION DISTANCES

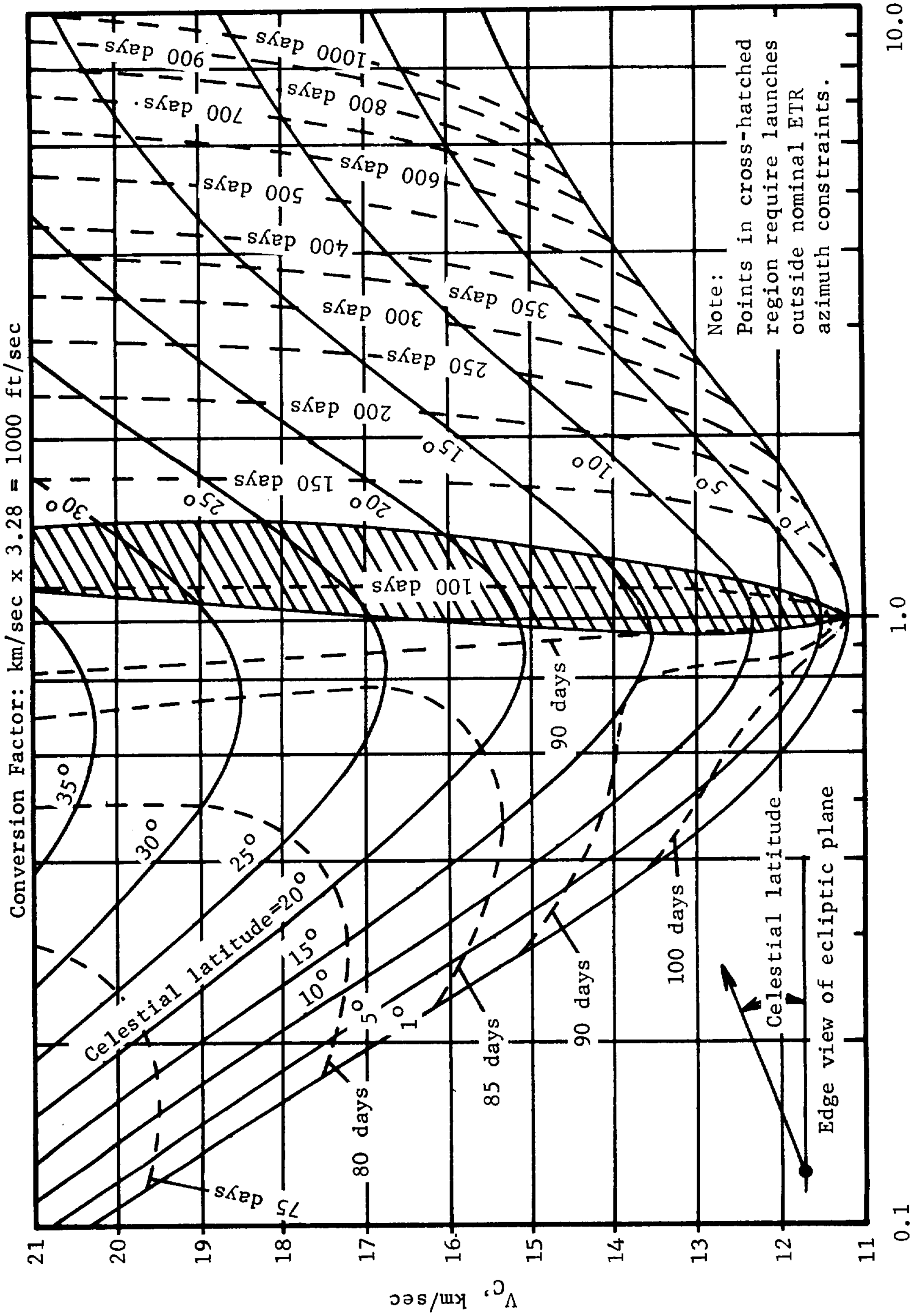


FIGURE 2-6. MINIMUM V_C AND CORRESPONDING FLIGHT TIMES FOR DIRECT OUT-OF-ECLIPTIC PROBES

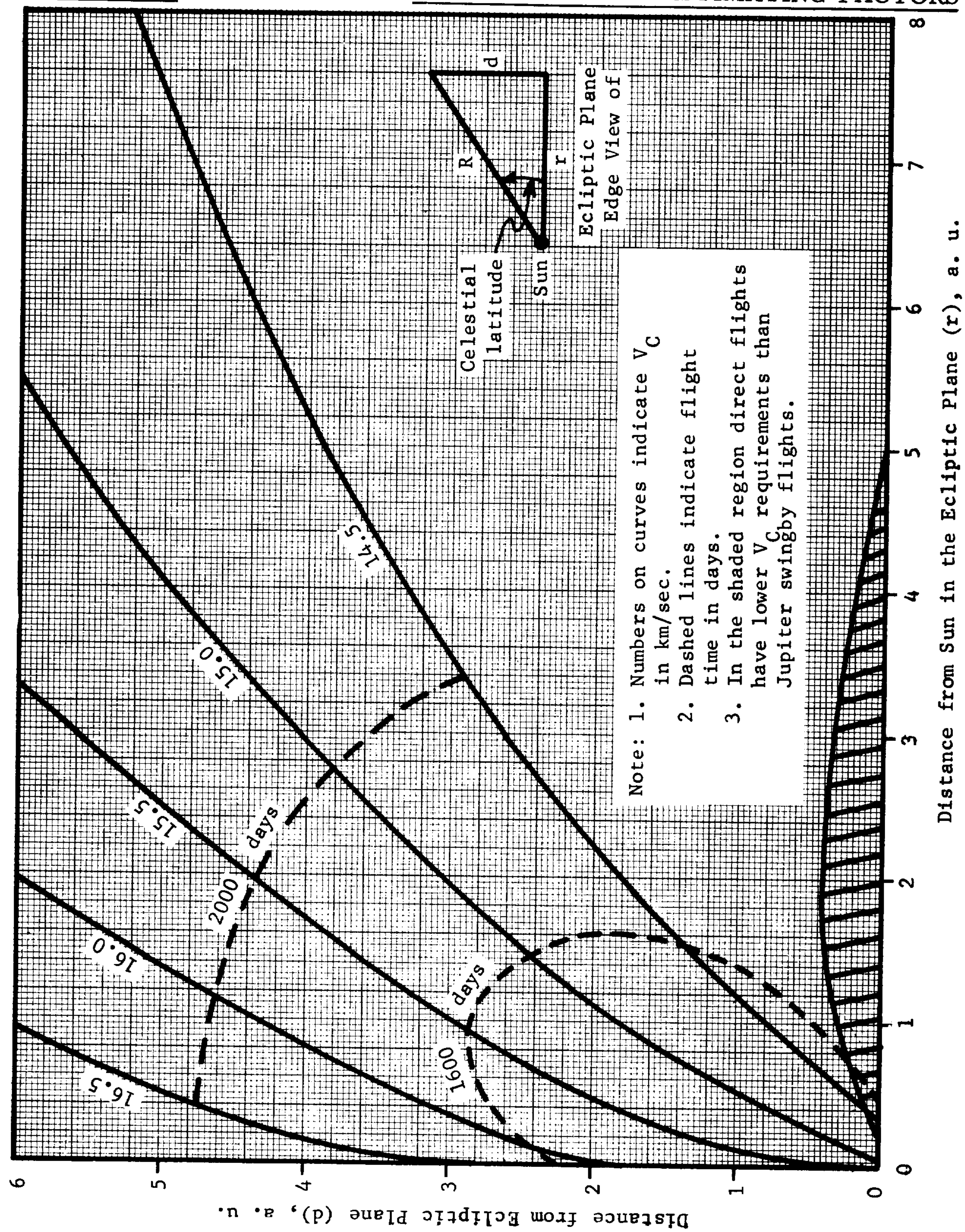


FIGURE 2-7. ACCESSIBLE REGION BOUNDARIES FOR OUT-OF-ECLIPTIC PROBES USING JUPITER SWINGBY

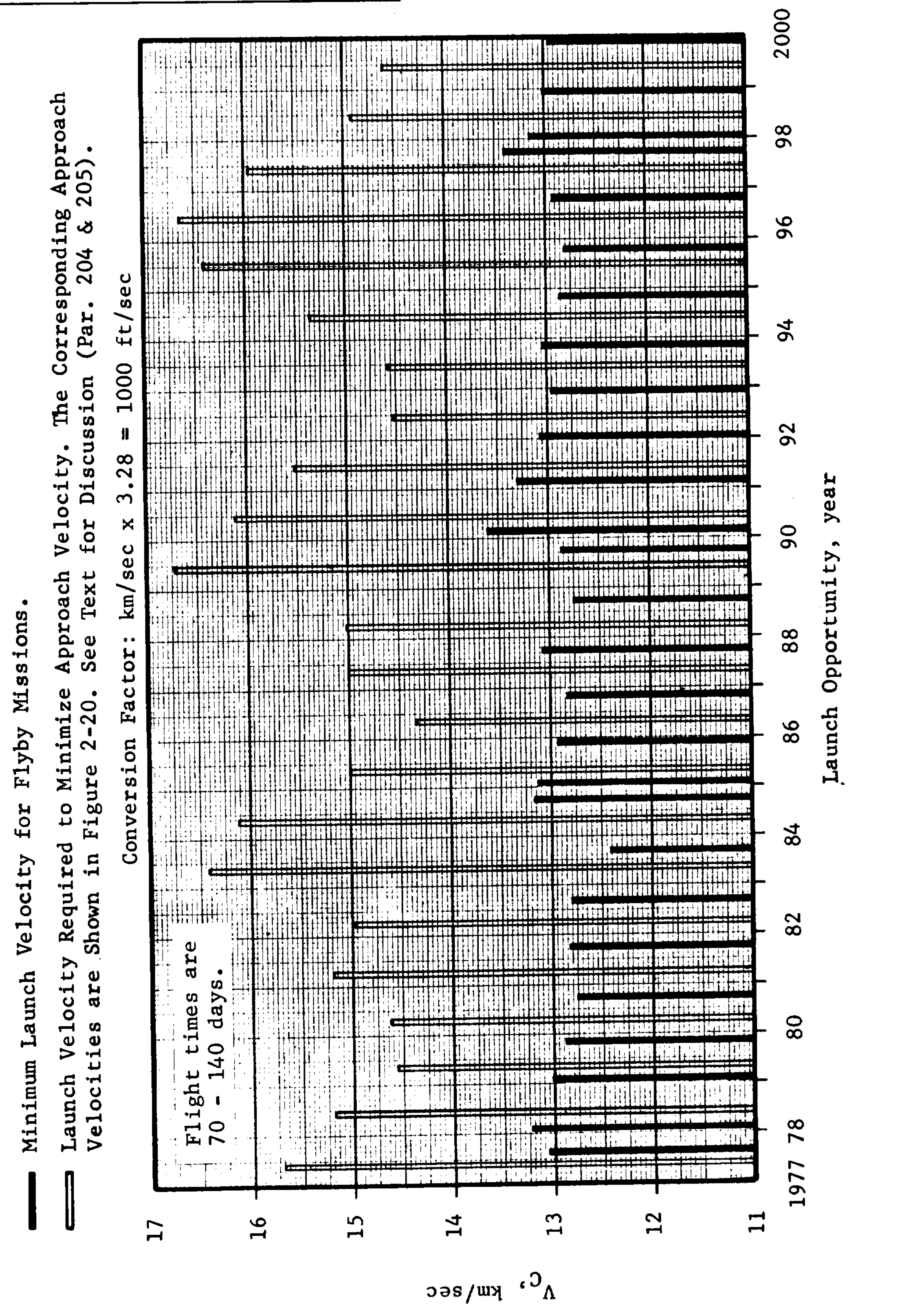


FIGURE 2-8. LAUNCH CHARACTERISTIC VELOCITY FOR DIRECT MERCURY MISSIONS

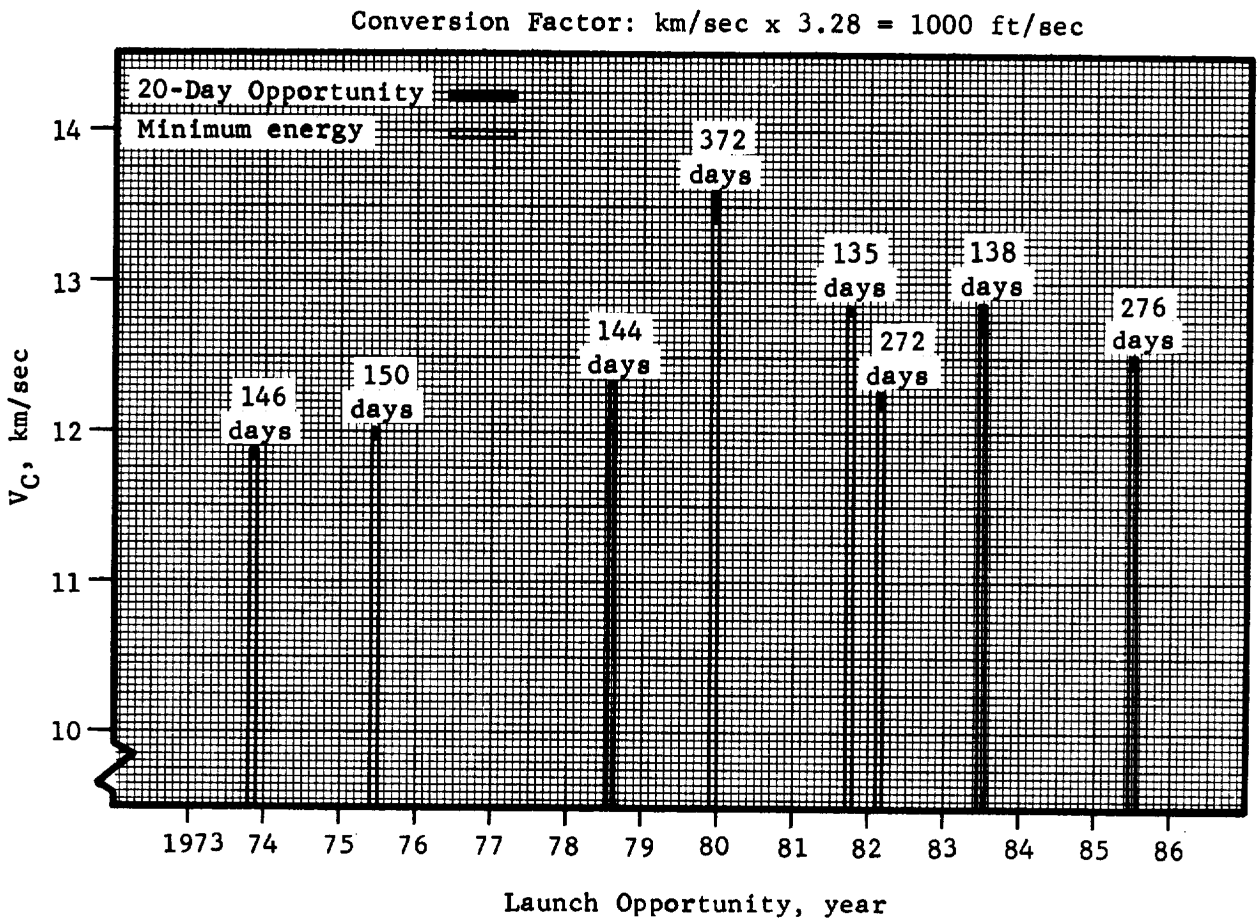


FIGURE 2-9. CHARACTERISTIC VELOCITY REQUIREMENTS FOR MERCURY MISSIONS USING VENUS SWINGBY

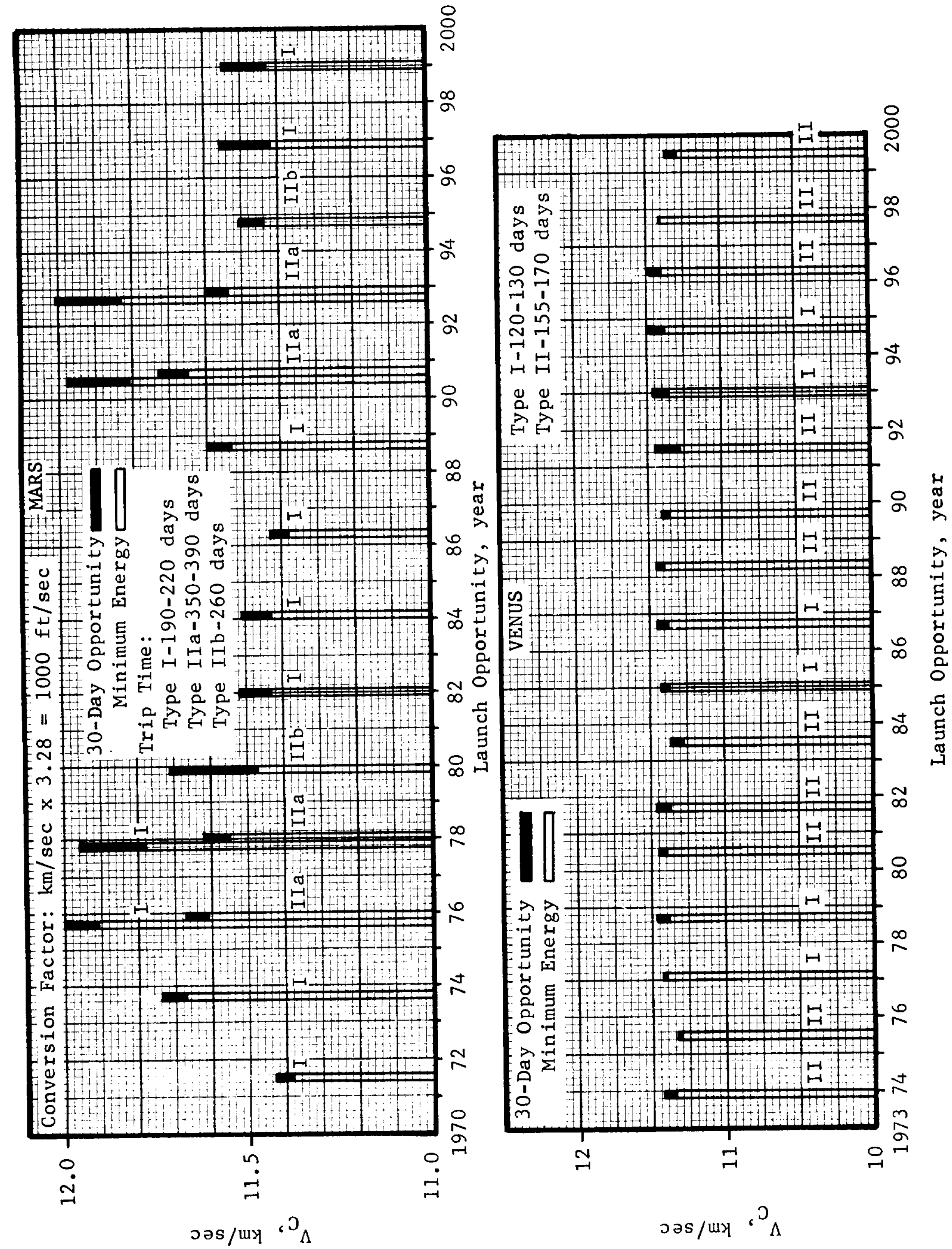


FIGURE 2-10. CHARACTERISTIC VELOCITY REQUIREMENTS FOR VENUS AND MARS

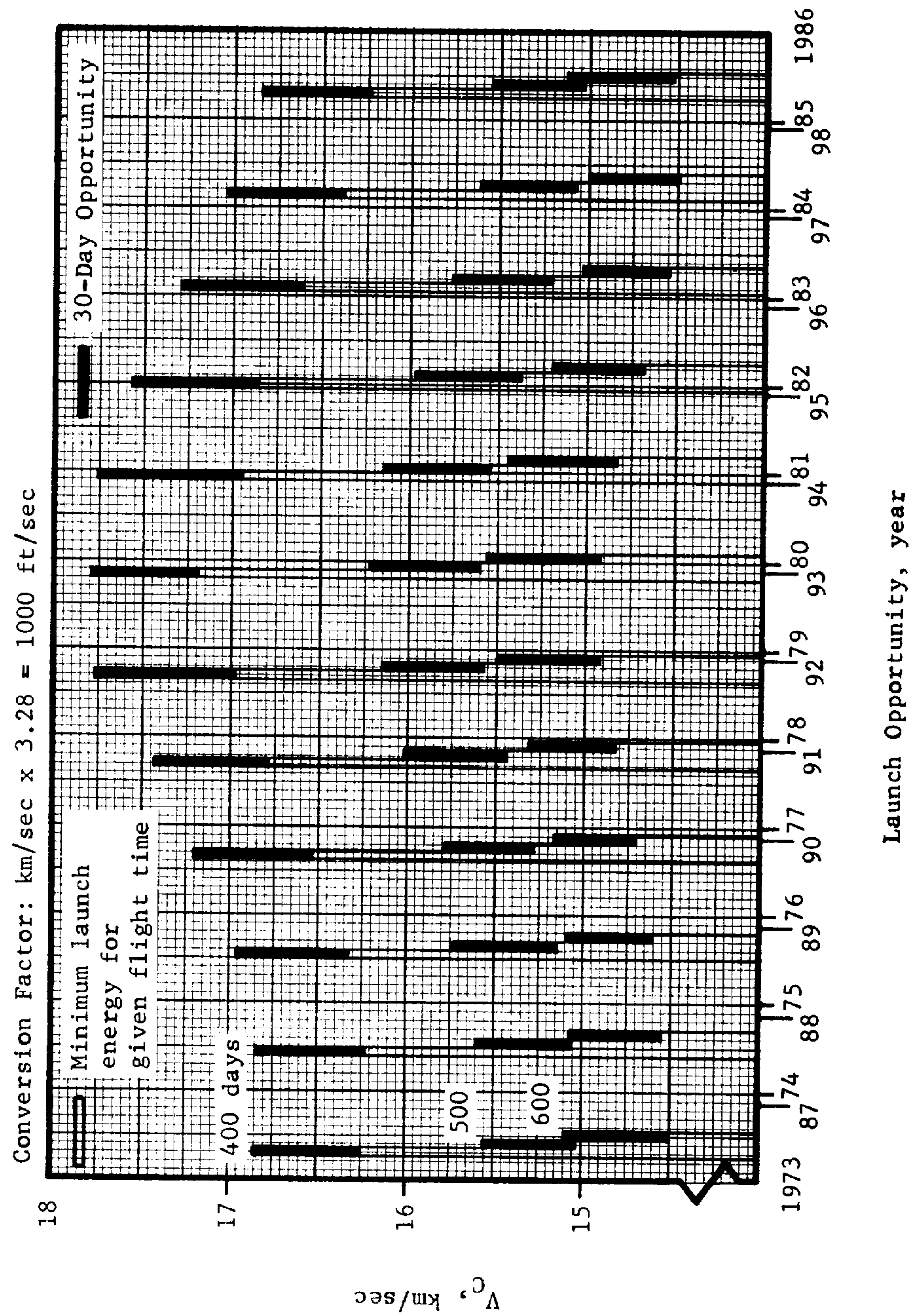


FIGURE 2-11. CHARACTERISTIC VELOCITY REQUIREMENTS FOR JUPITER (400, 500, AND 600 DAY FLIGHT TIMES)

Notes: a. Minimum energy for 1625 day flight.
b. 30 day opportunity for minimum energy trajectory, frequently with shorter flight time.

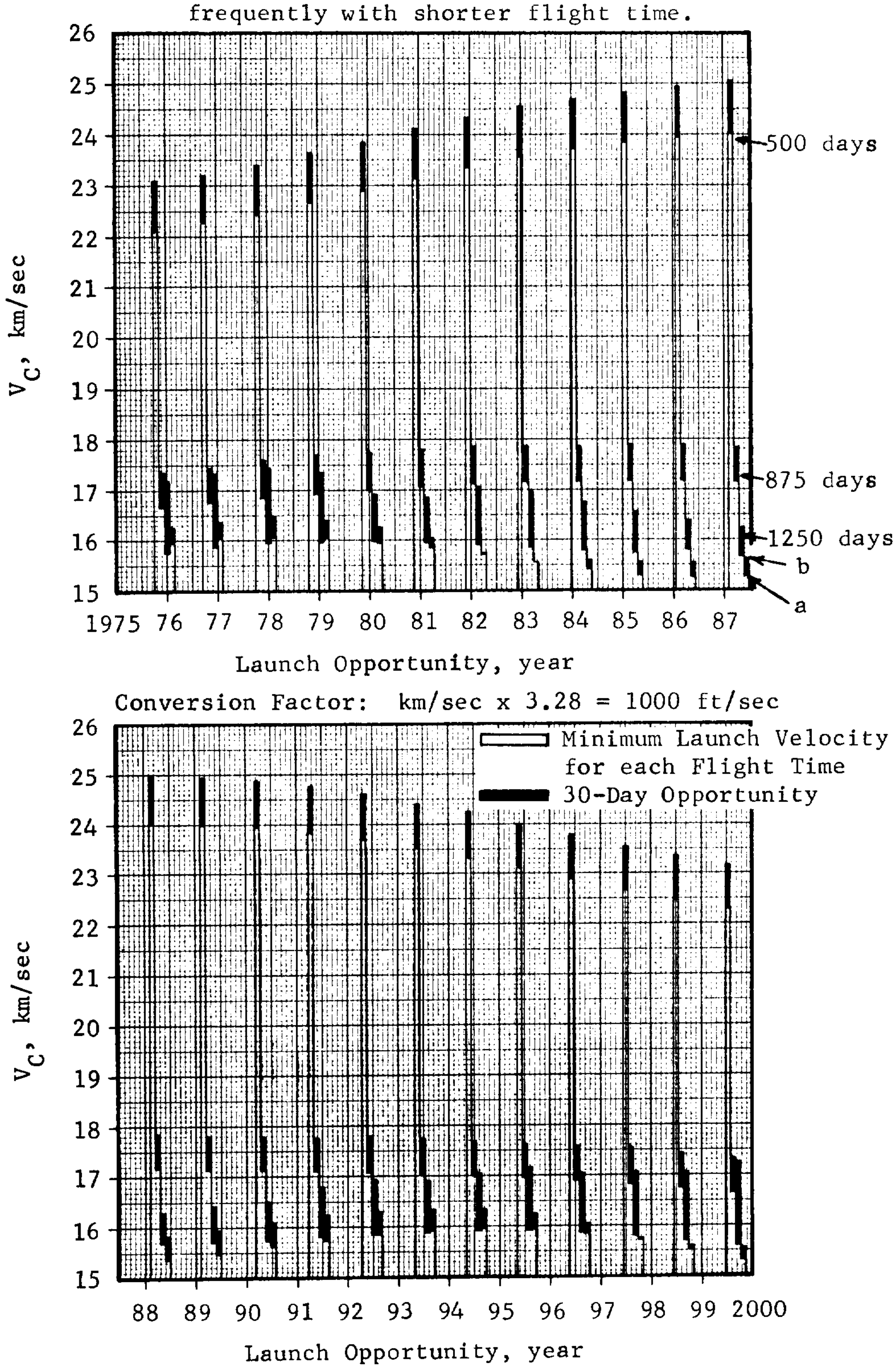


FIGURE 2-12. CHARACTERISTIC VELOCITY REQUIREMENTS FOR SATURN (500, 875, 1250, AND 1625 DAY FLIGHT TIMES)

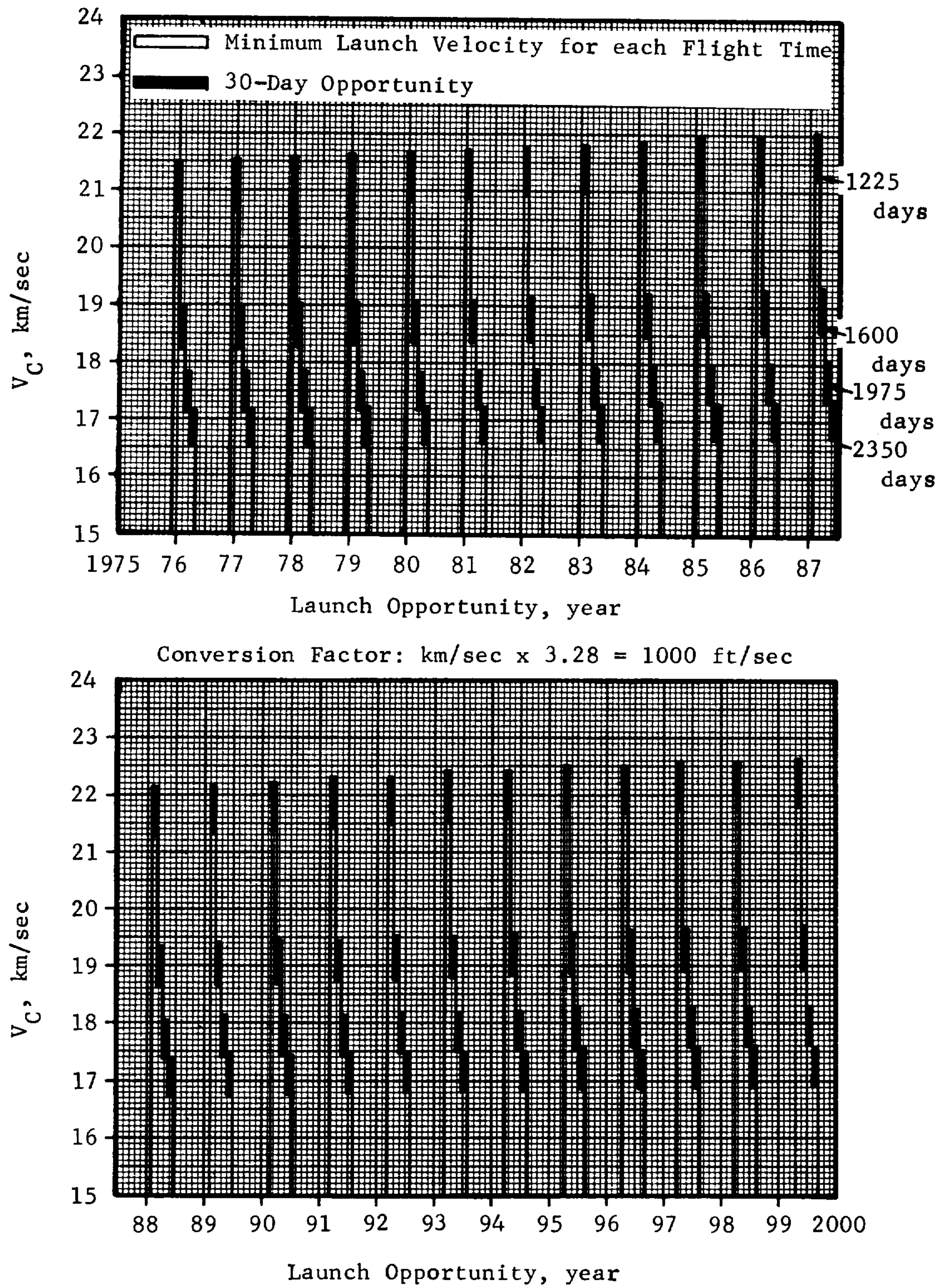


FIGURE 2-13. CHARACTERISTIC VELOCITY REQUIREMENTS FOR URANUS (1225, 1600, 1975, AND 2350 DAY FLIGHT TIMES)

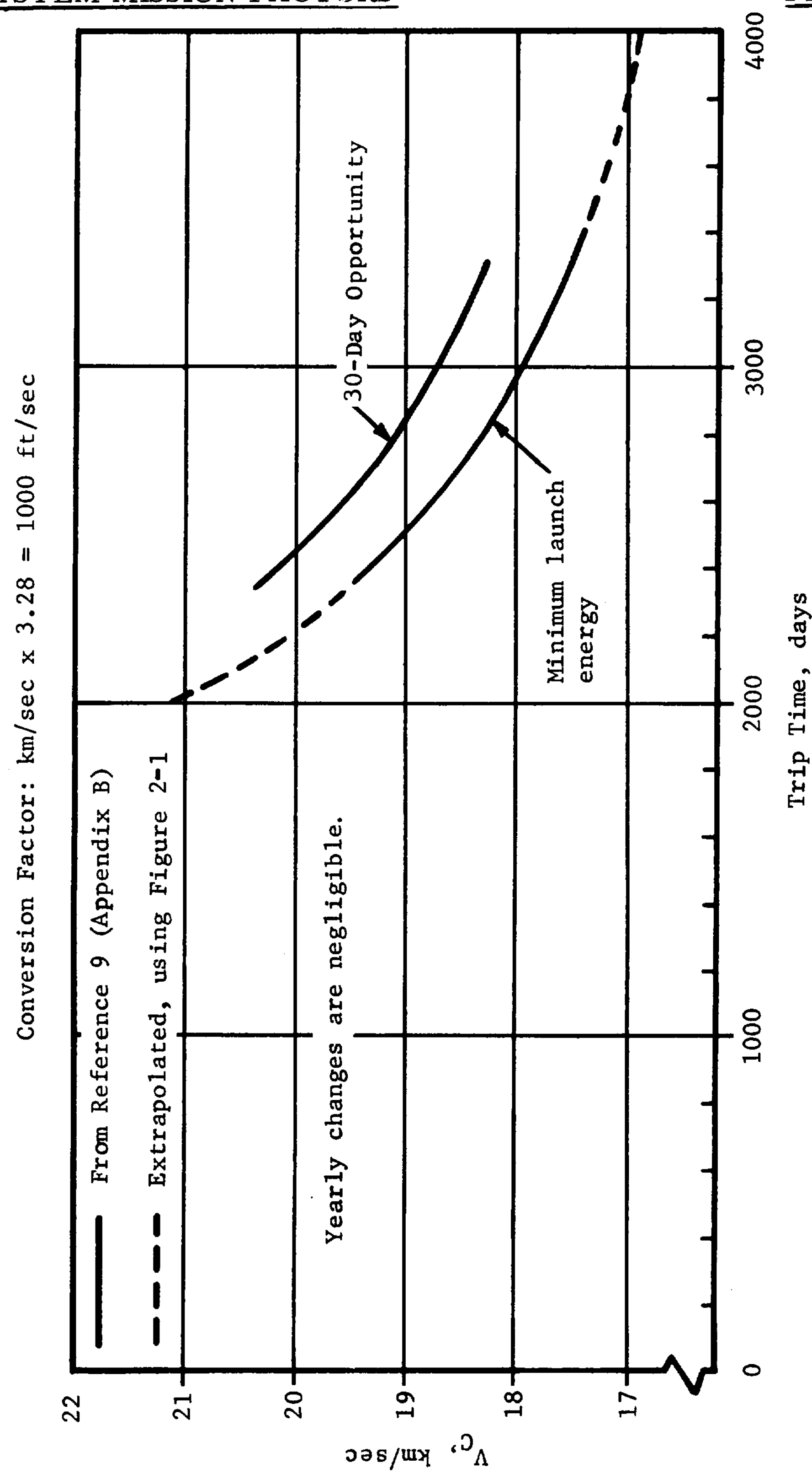


FIGURE 2-14. CHARACTERISTIC VELOCITY REQUIREMENTS FOR NEPTUNE

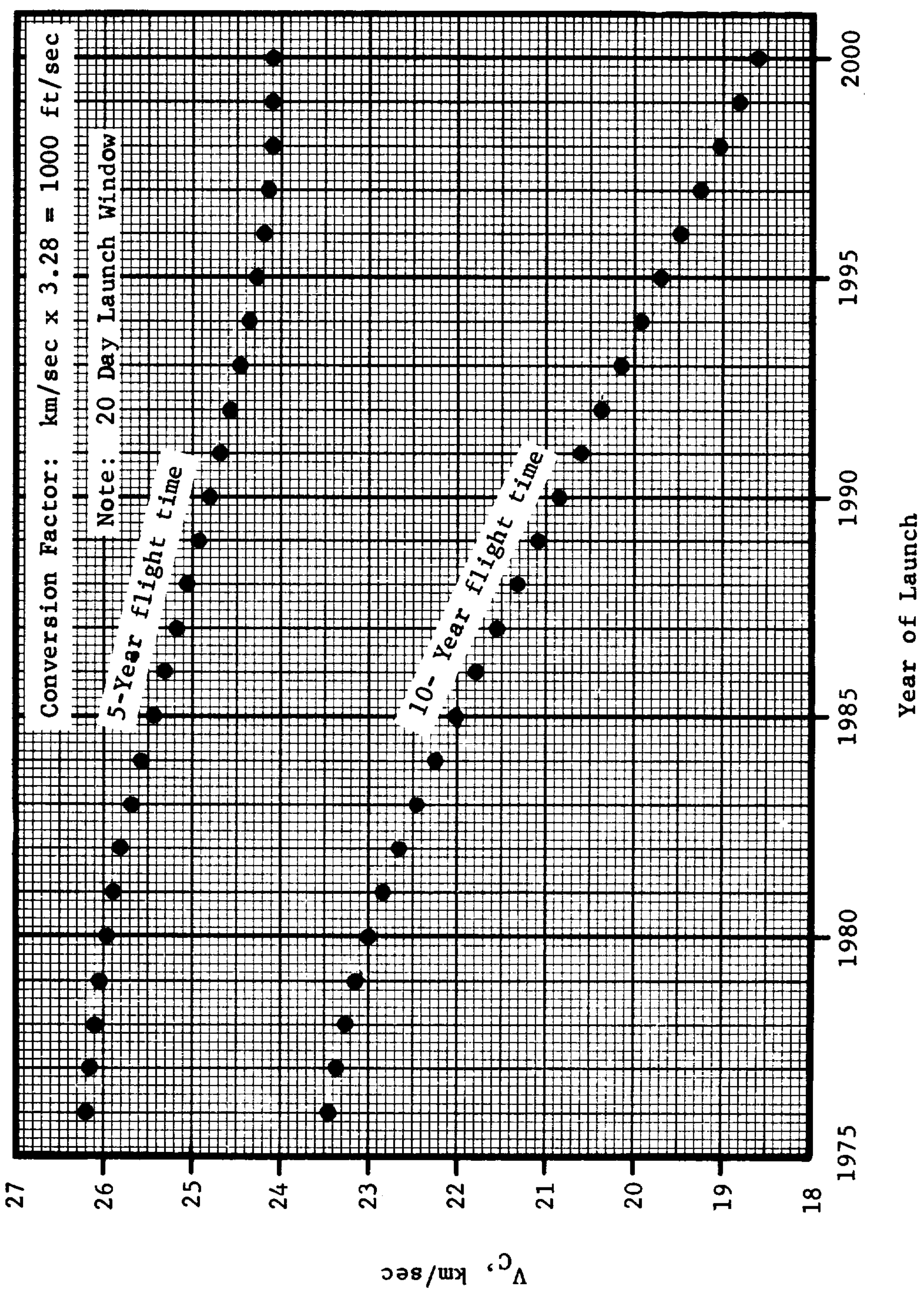


FIGURE 2-15. CHARACTERISTIC VELOCITY REQUIREMENTS FOR PLUTO

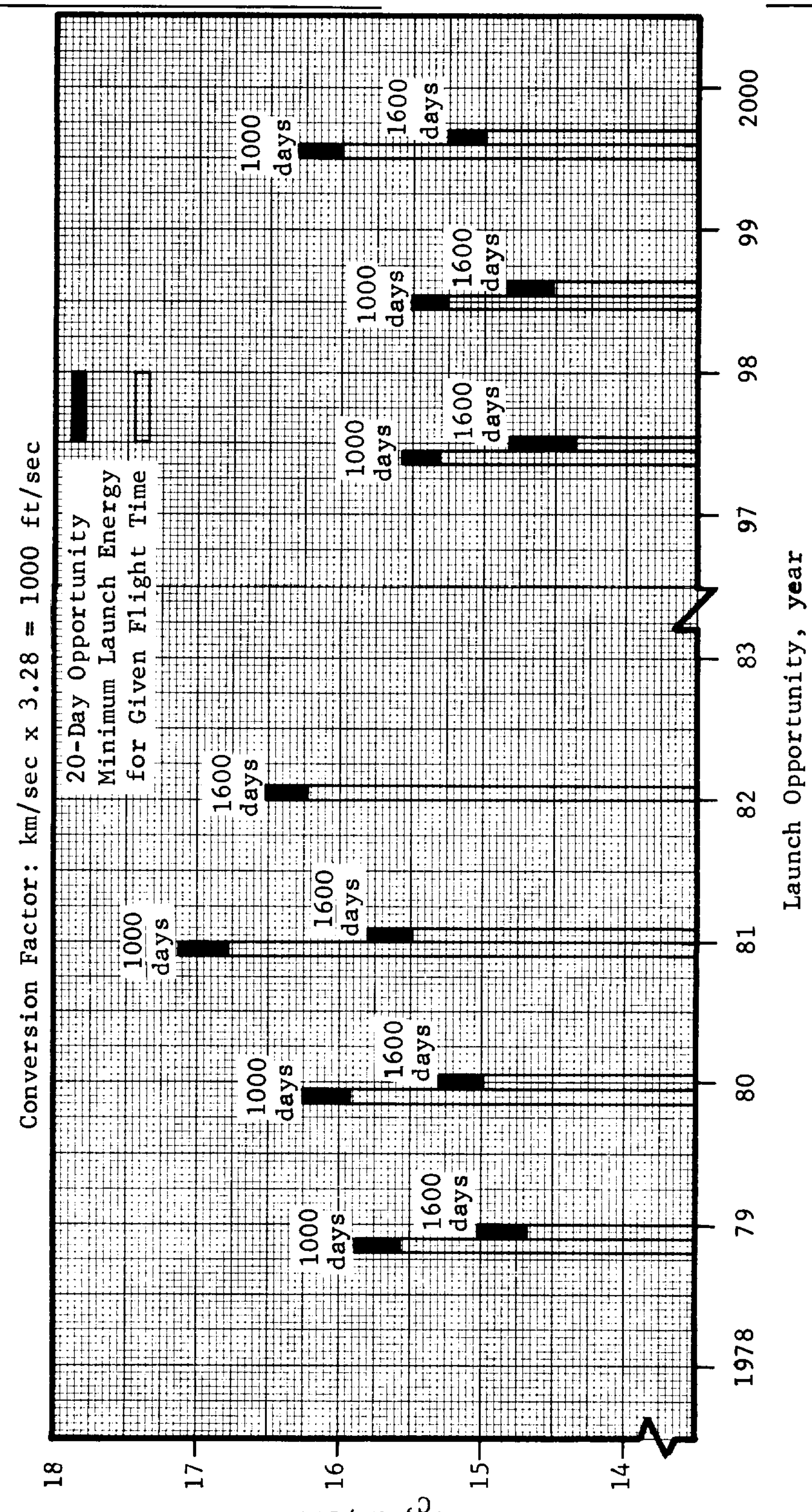


FIGURE 2-16a. CHARACTERISTIC VELOCITY REQUIREMENTS FOR JUPITER SWINGBYS TO SATURN

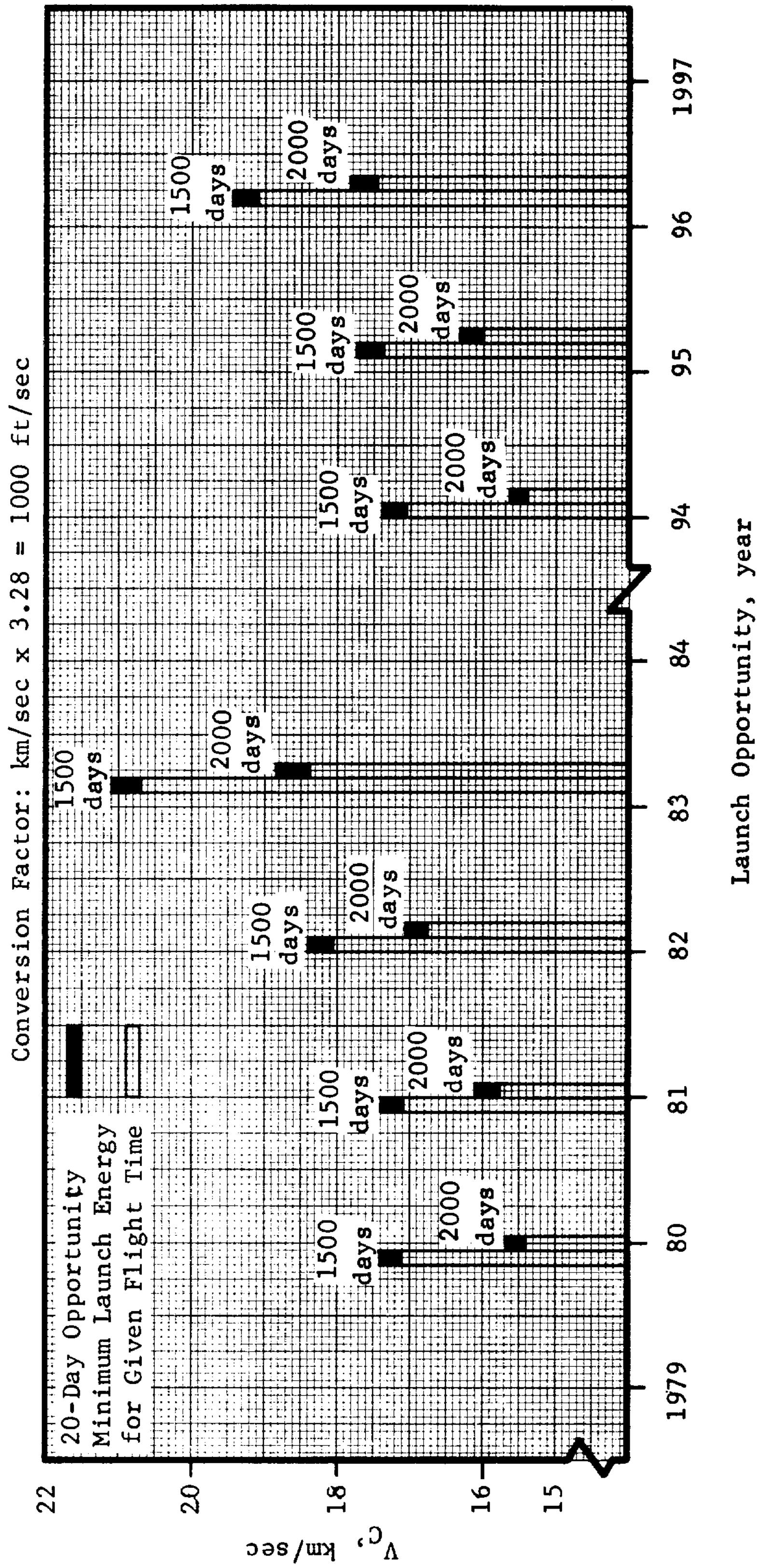


FIGURE 2-16b. CHARACTERISTIC VELOCITY REQUIREMENTS FOR JUPITER SWINGBYS TO URANUS

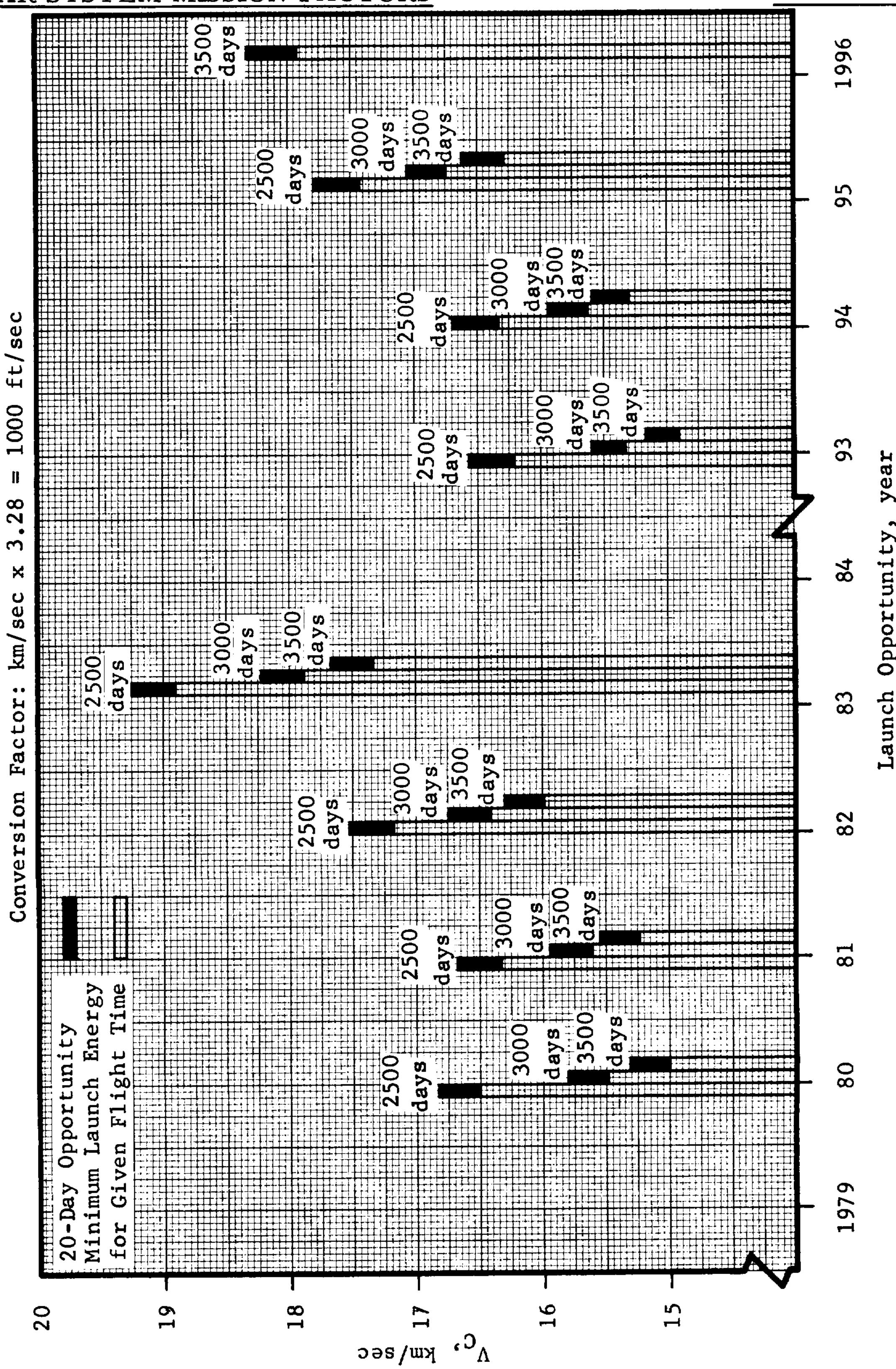


FIGURE 2-16c. CHARACTERISTIC VELOCITY REQUIREMENTS FOR JUPITER SWINGBYS TO NEPTUNE

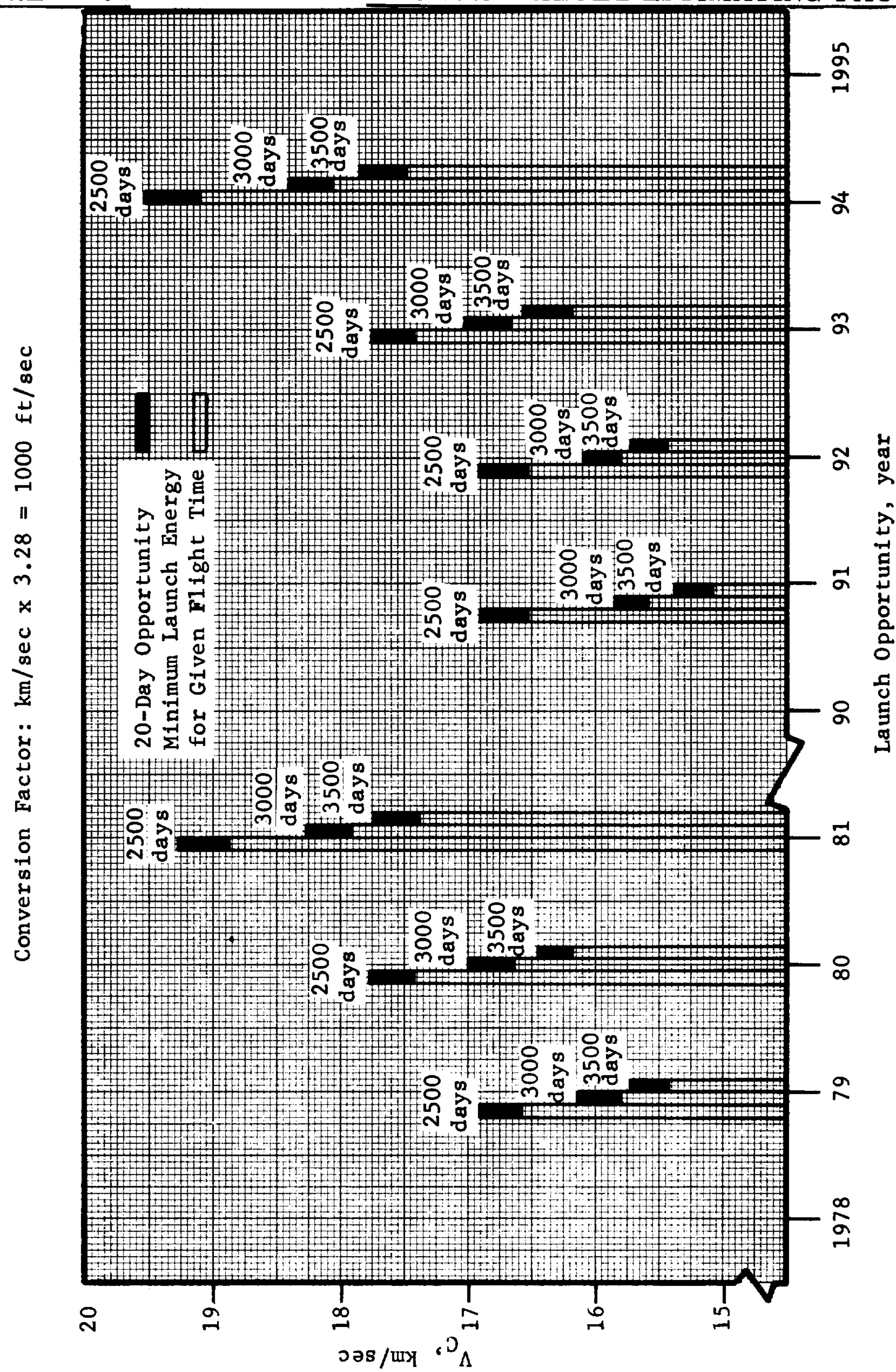
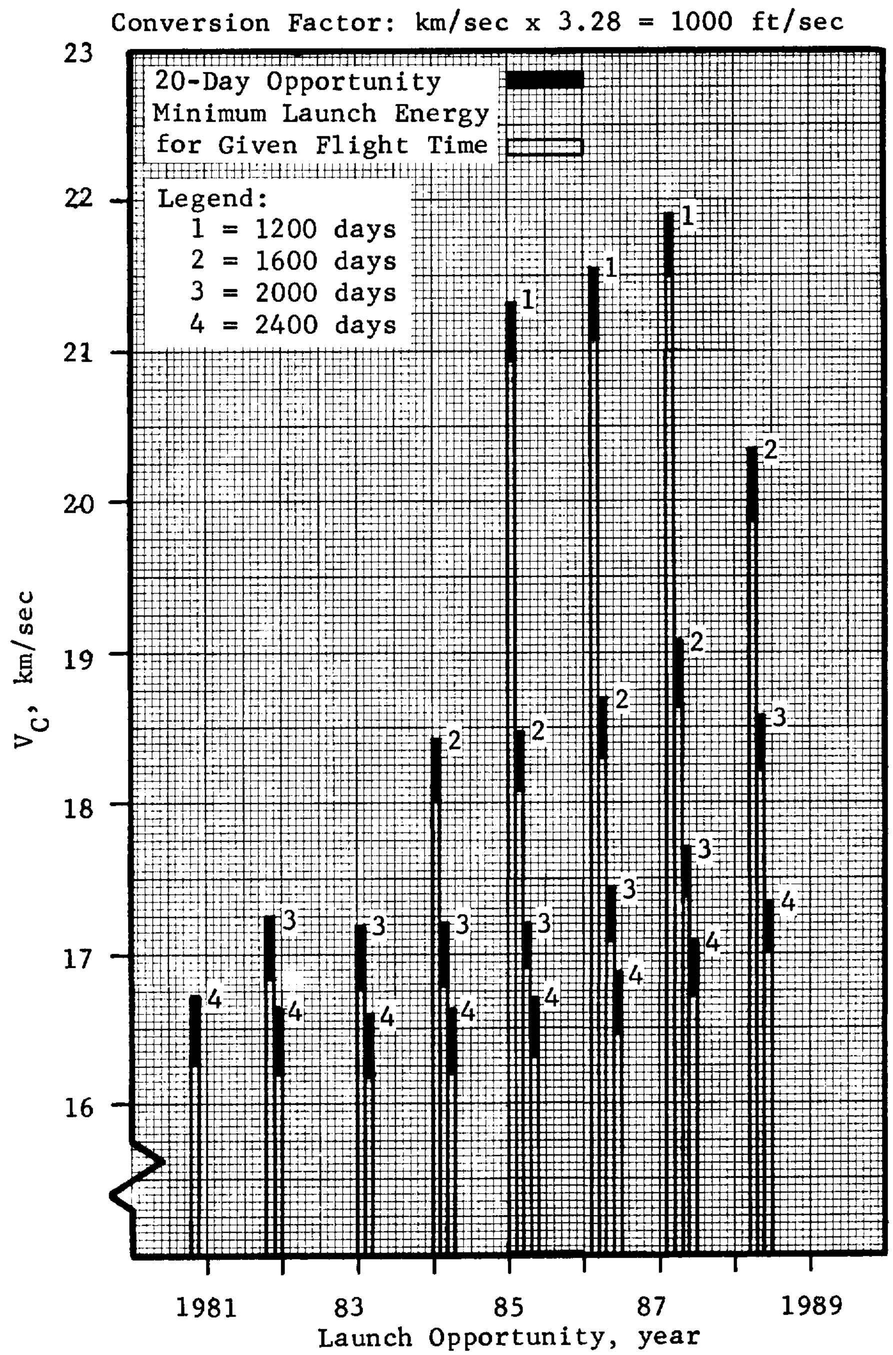


FIGURE 2-16d. CHARACTERISTIC VELOCITY REQUIREMENTS FOR JUPITER SWINGBYS TO PLUTO



Note: The short flight times are not feasible in some years due to planetary alignment or a need to pass too close or through the swingby planet.

FIGURE 2-17a. CHARACTERISTIC VELOCITY REQUIREMENTS FOR SWINGBY MISSIONS TO URANUS VIA SATURN

Conversion Factor:
km/sec x 3.28 = 1000 ft/sec

Note: The short flight times are not feasible in some years due to planetary alignment or a need to pass too close or through the swingby planet.

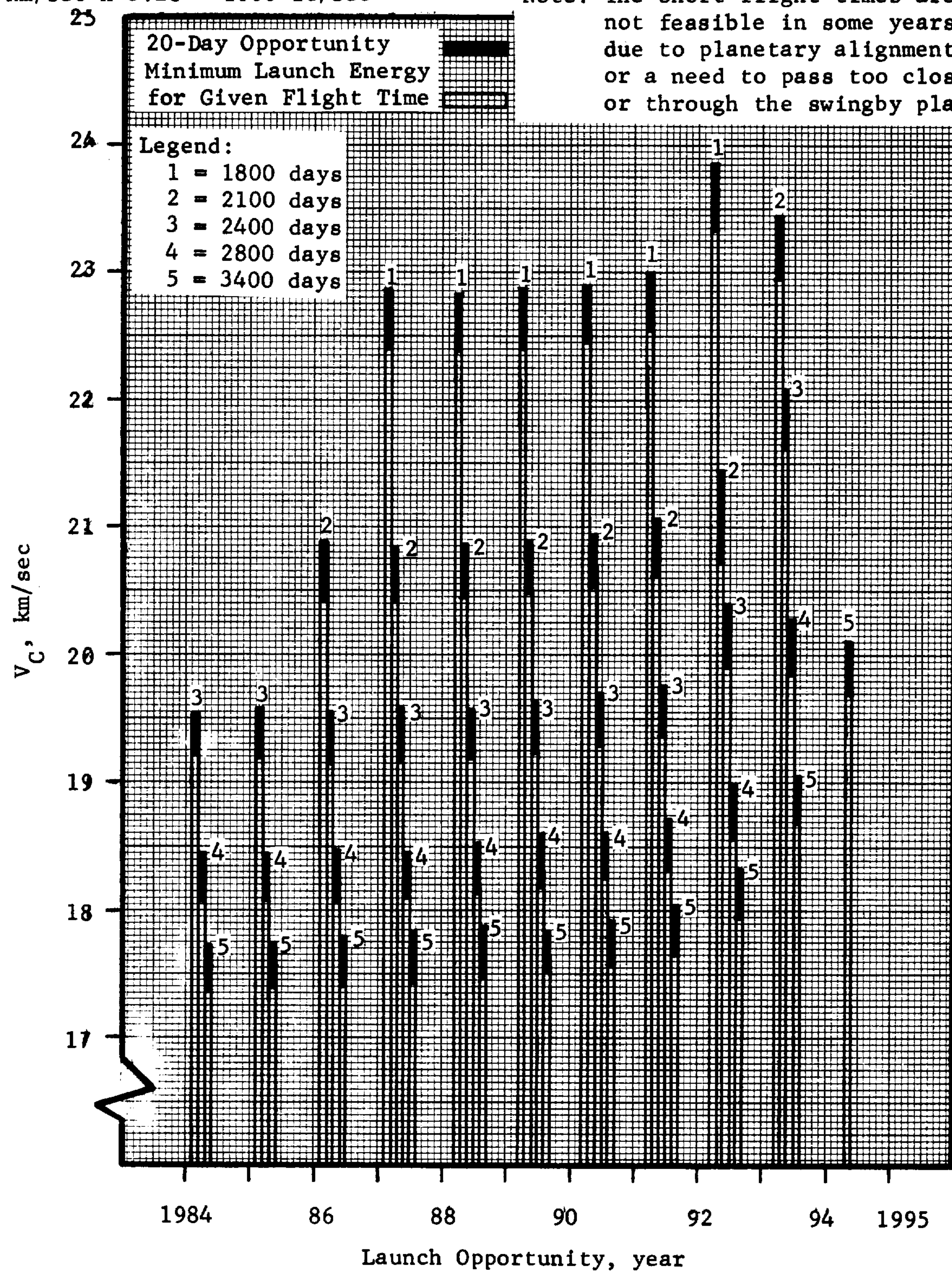


FIGURE 2-17b. CHARACTERISTIC VELOCITY REQUIREMENTS FOR SWINGBY MISSIONS TO NEPTUNE VIA URANUS

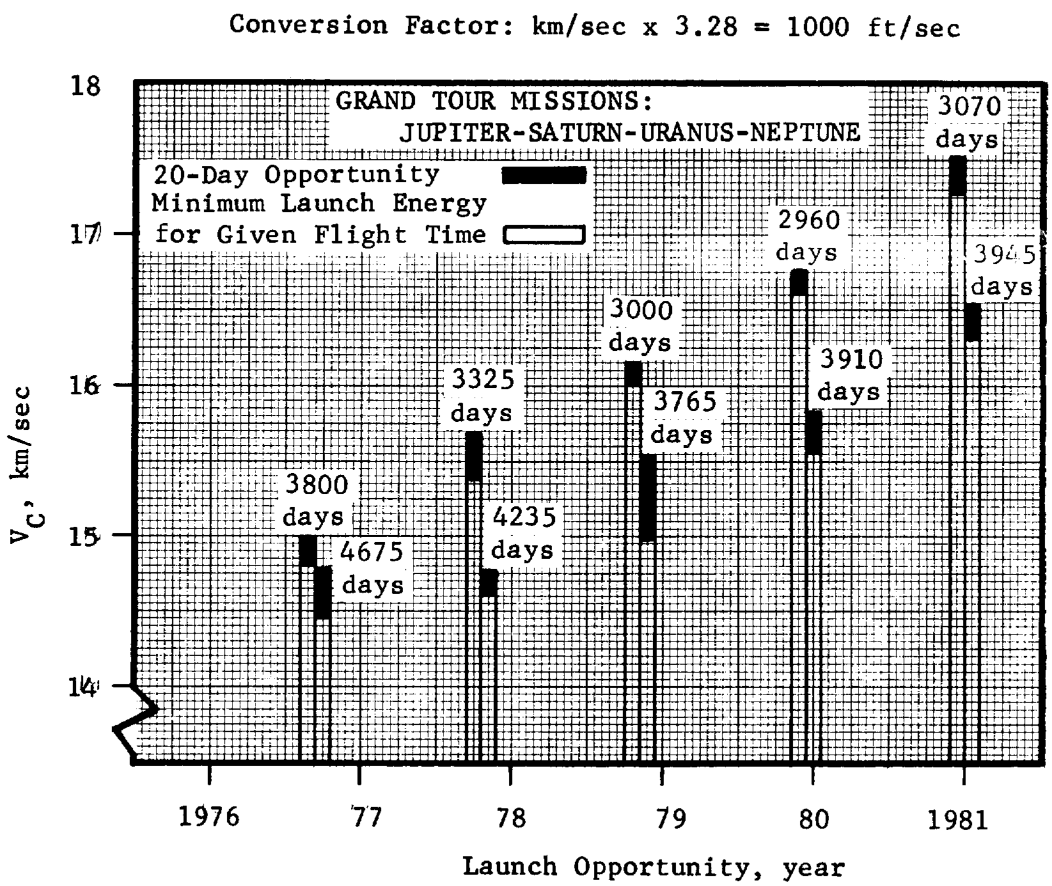


FIGURE 2-18. CHARACTERISTIC VELOCITY REQUIREMENTS FOR MULTI-PLANET FLYBYS

Conversion Factor: km/sec x 3.28 = 1000 ft/sec

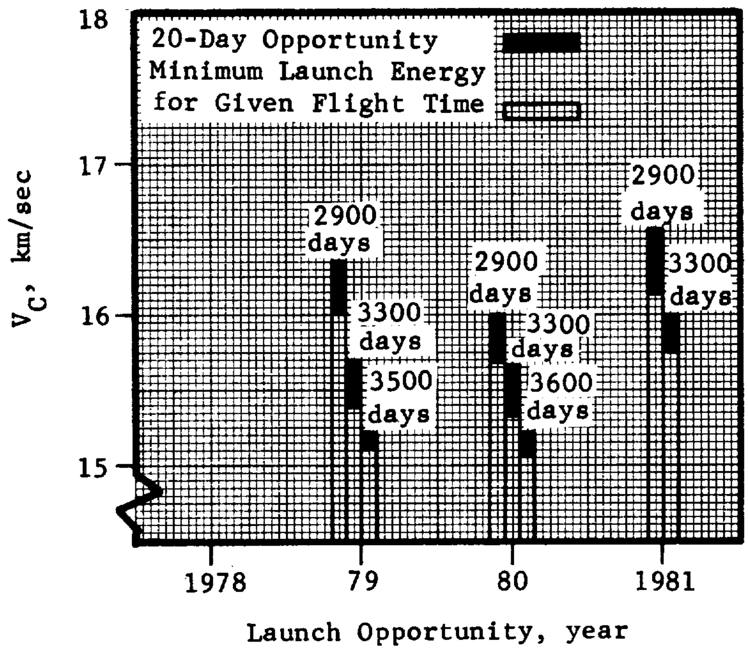


FIGURE 2-19a. CHARACTERISTIC VELOCITY REQUIREMENTS FOR MULTI-PLANET FLYBYS (JUPITER-URANUS-NEPTUNE)

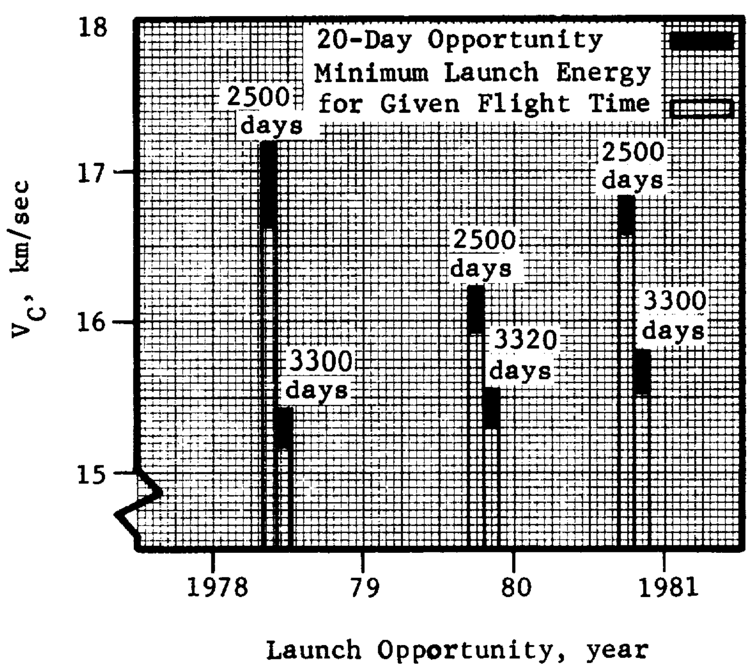


FIGURE 2-19b. CHARACTERISTIC VELOCITY REQUIREMENTS FOR MULTI-PLANET FLYBYS (JUPITER-SATURN-PLUTO)

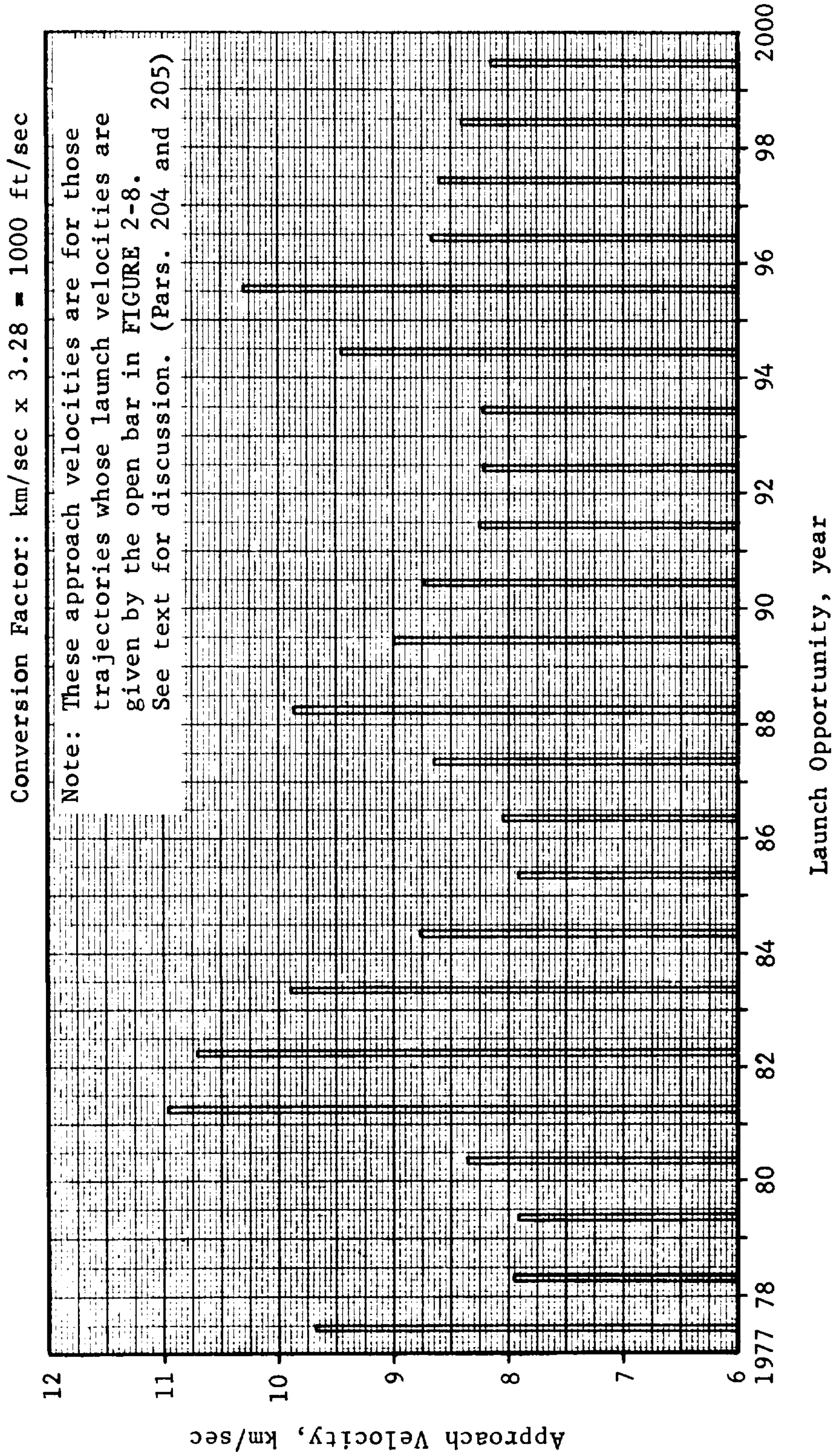


FIGURE 2-20. MINIMUM APPROACH VELOCITIES FOR DIRECT MERCURY FLIGHTS

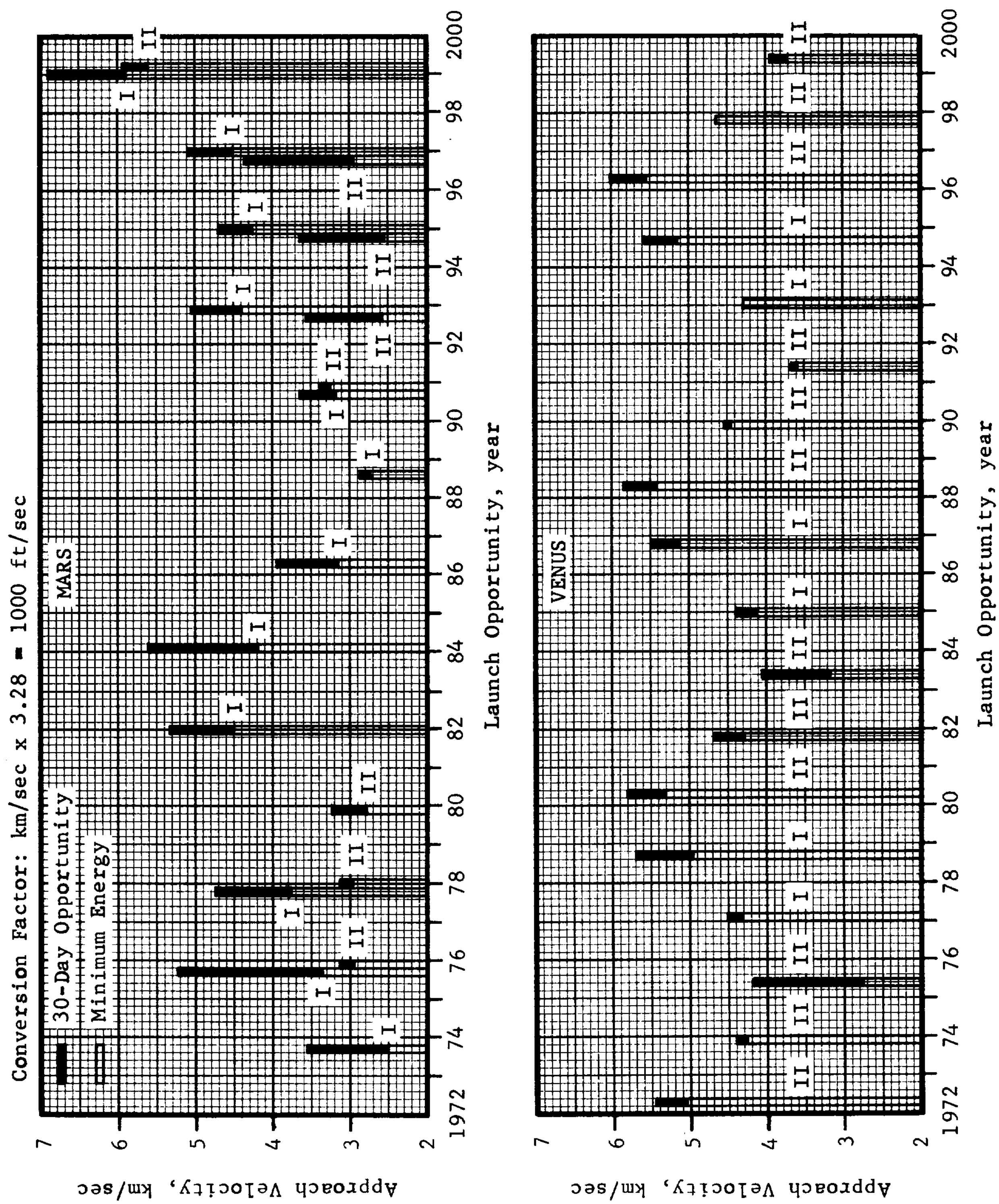


FIGURE 2-21. APPROACH VELOCITY FOR MARS AND VENUS

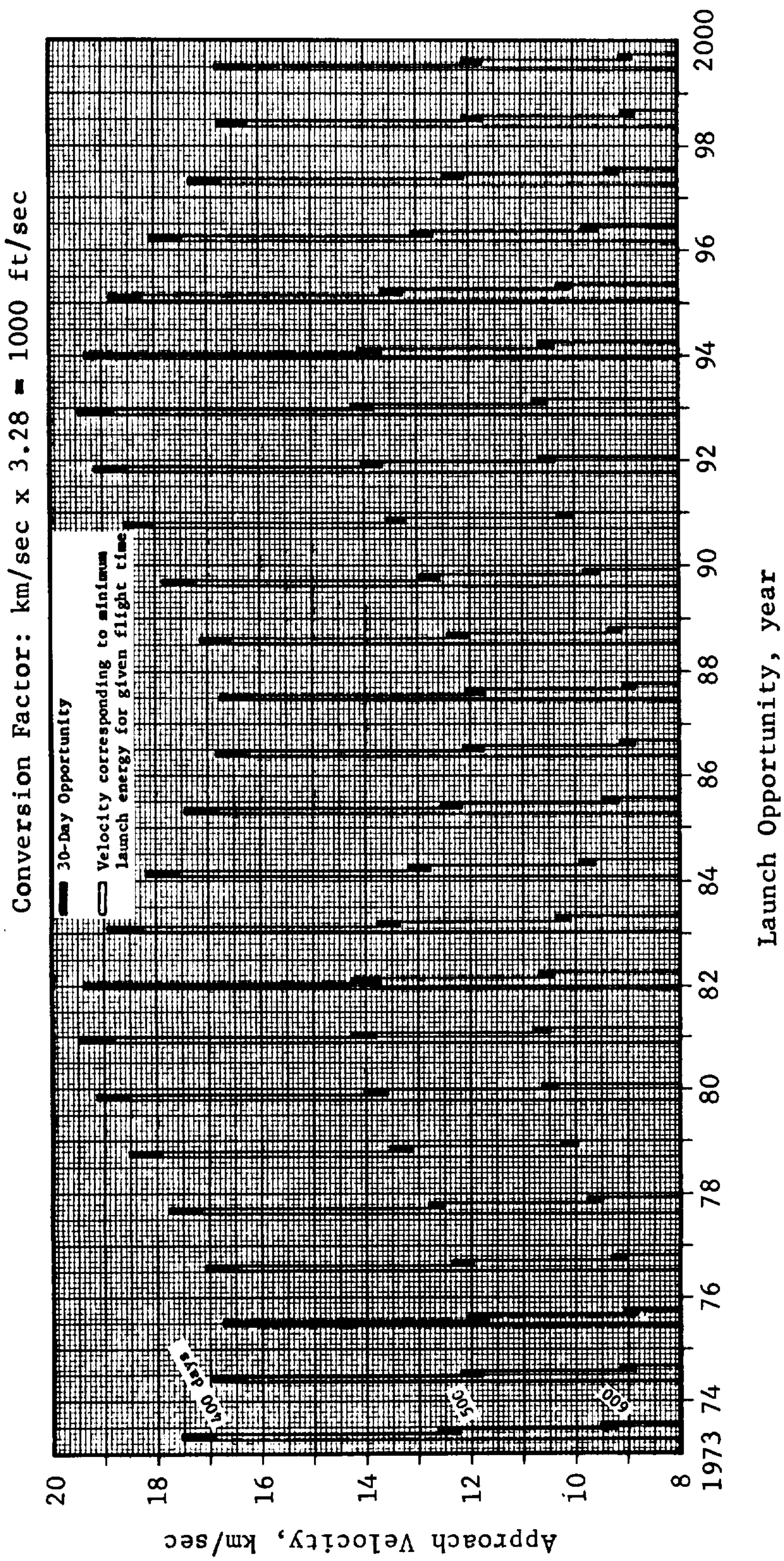


FIGURE 2-22. APPROACH VELOCITY FOR JUPITER (400, 500, AND 600 DAY FLIGHT TIMES)

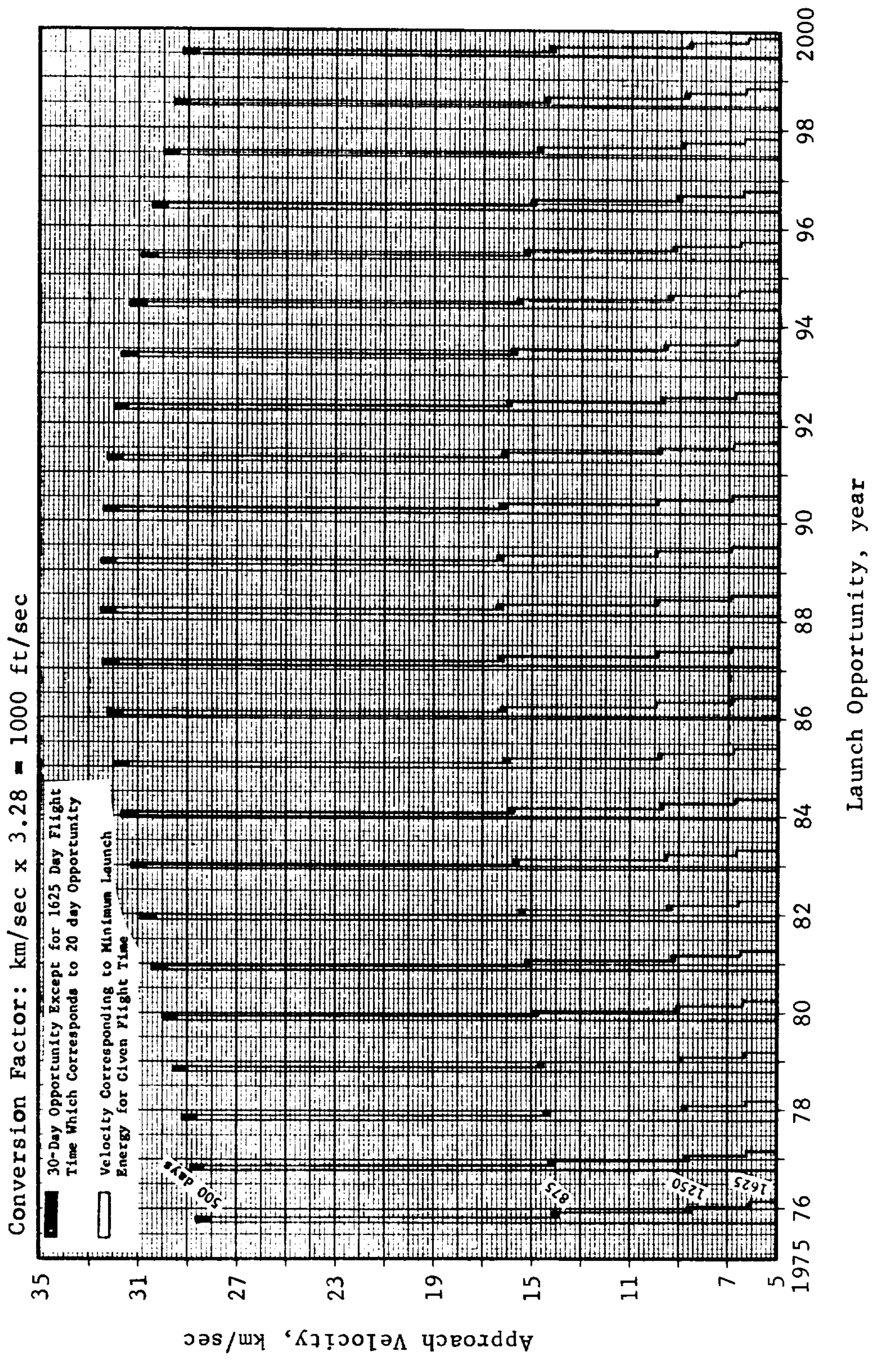


FIGURE 2-23. APPROACH VELOCITY FOR SATURN (500, 875, 1250, AND 1625 DAY FLIGHT TIMES)

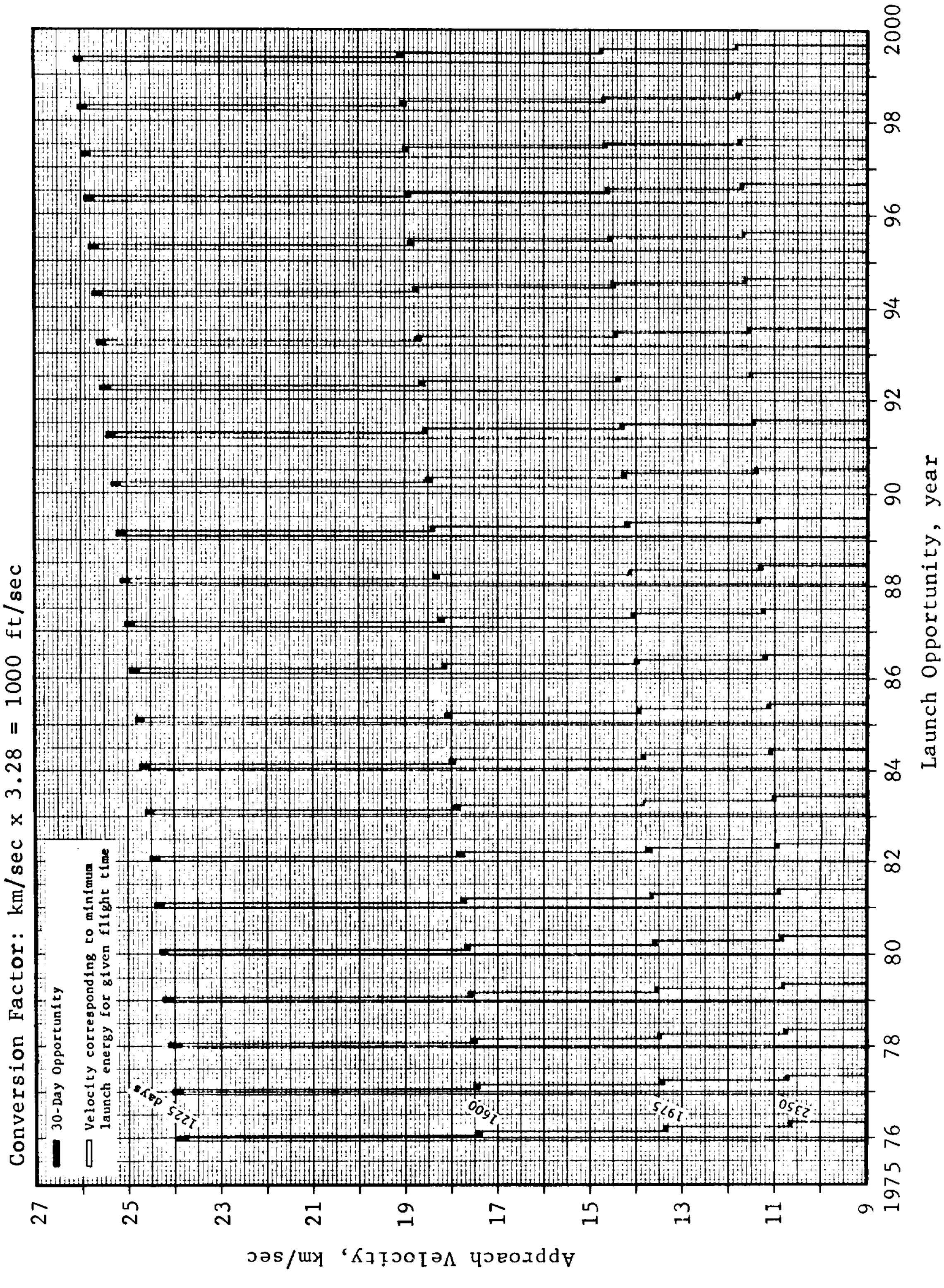


FIGURE 2-24. APPROACH VELOCITY FOR URANUS (1225, 1600, 1975, AND 2350 DAY FLIGHT TIMES)

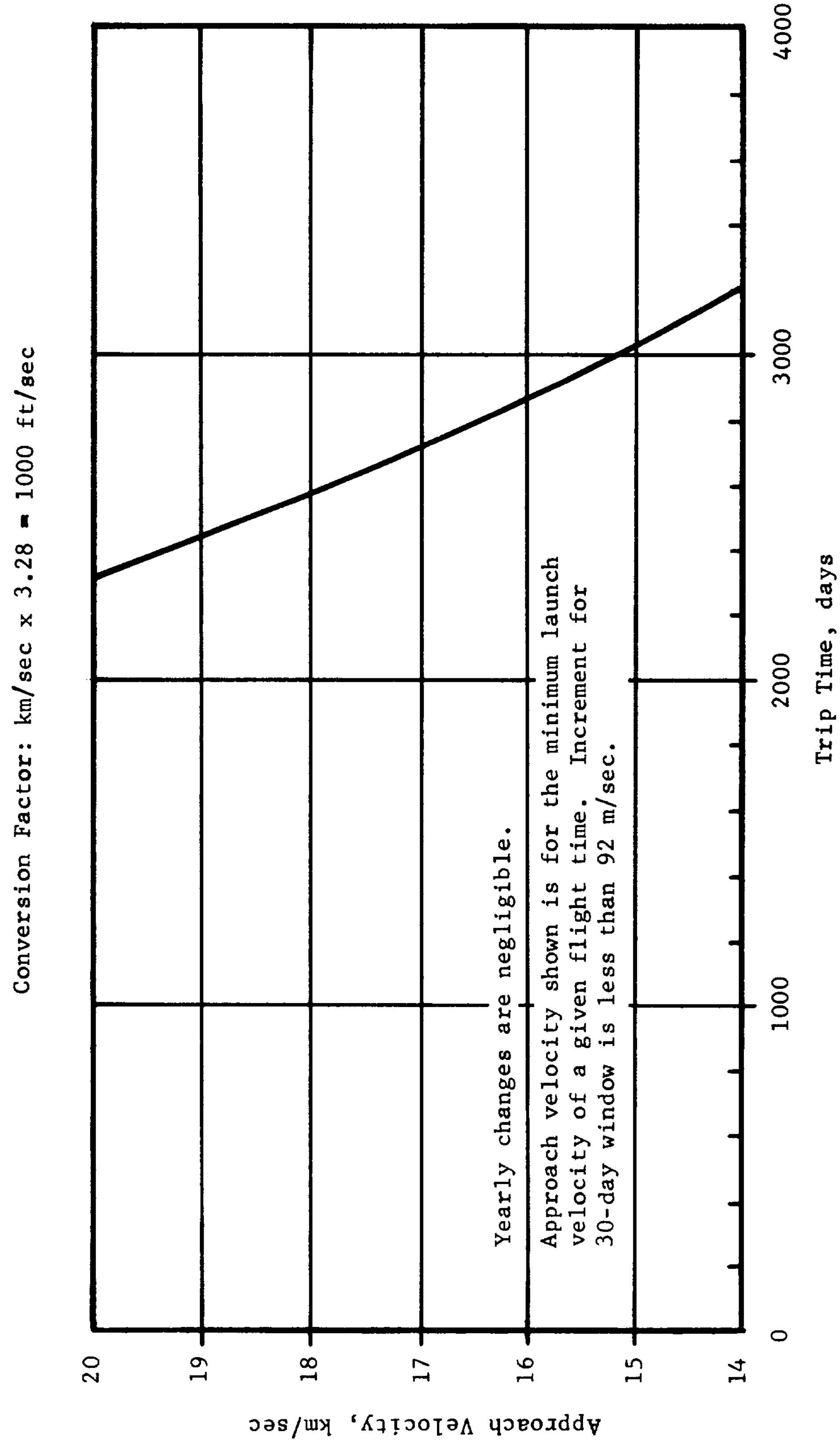


FIGURE 2-25. APPROACH VELOCITY FOR NEPTUNE

Conversion Factor: km/sec x 3.28 = 1000 ft/sec

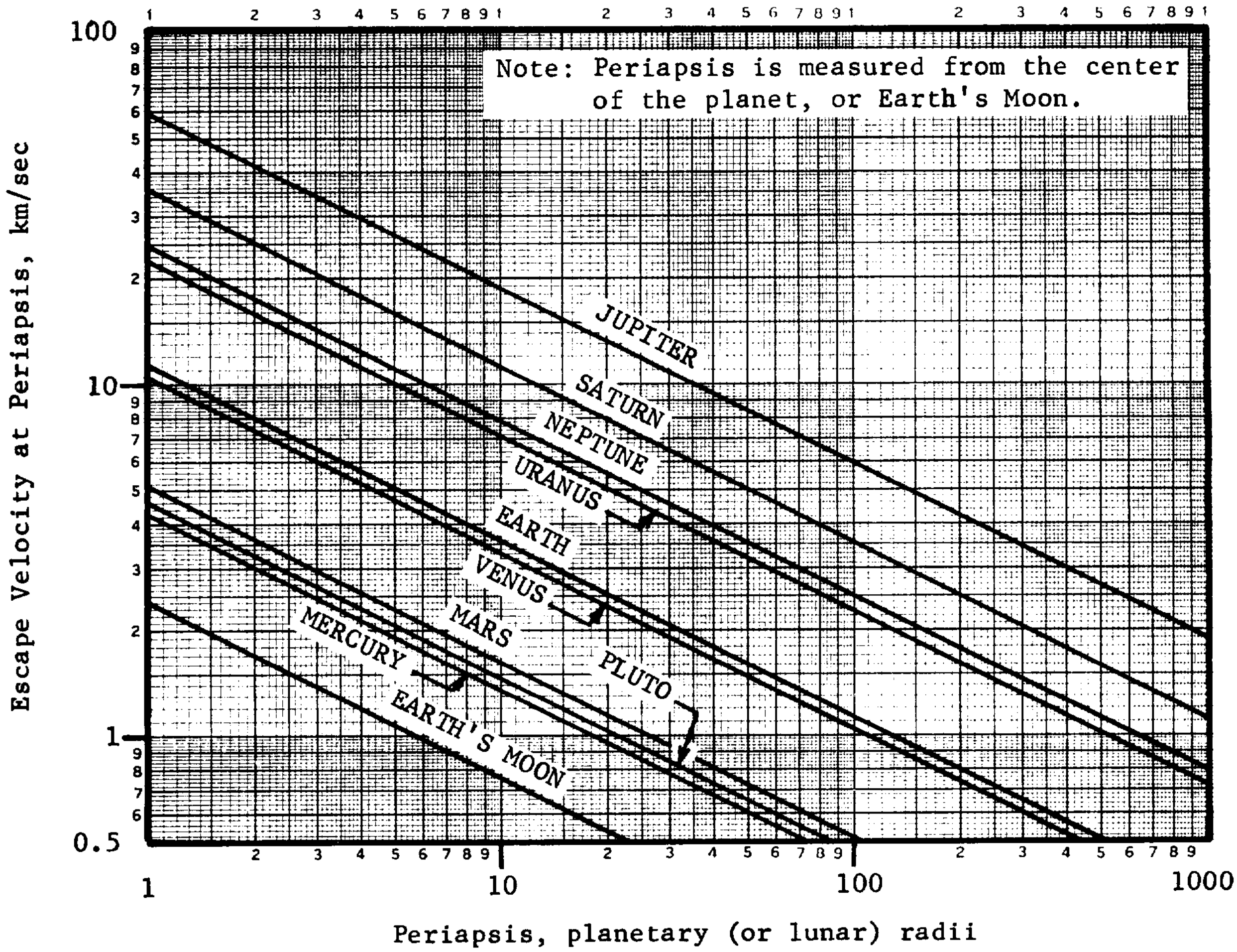


FIGURE 2-26. ESCAPE VELOCITIES FOR THE PLANETS AND EARTH'S MOON

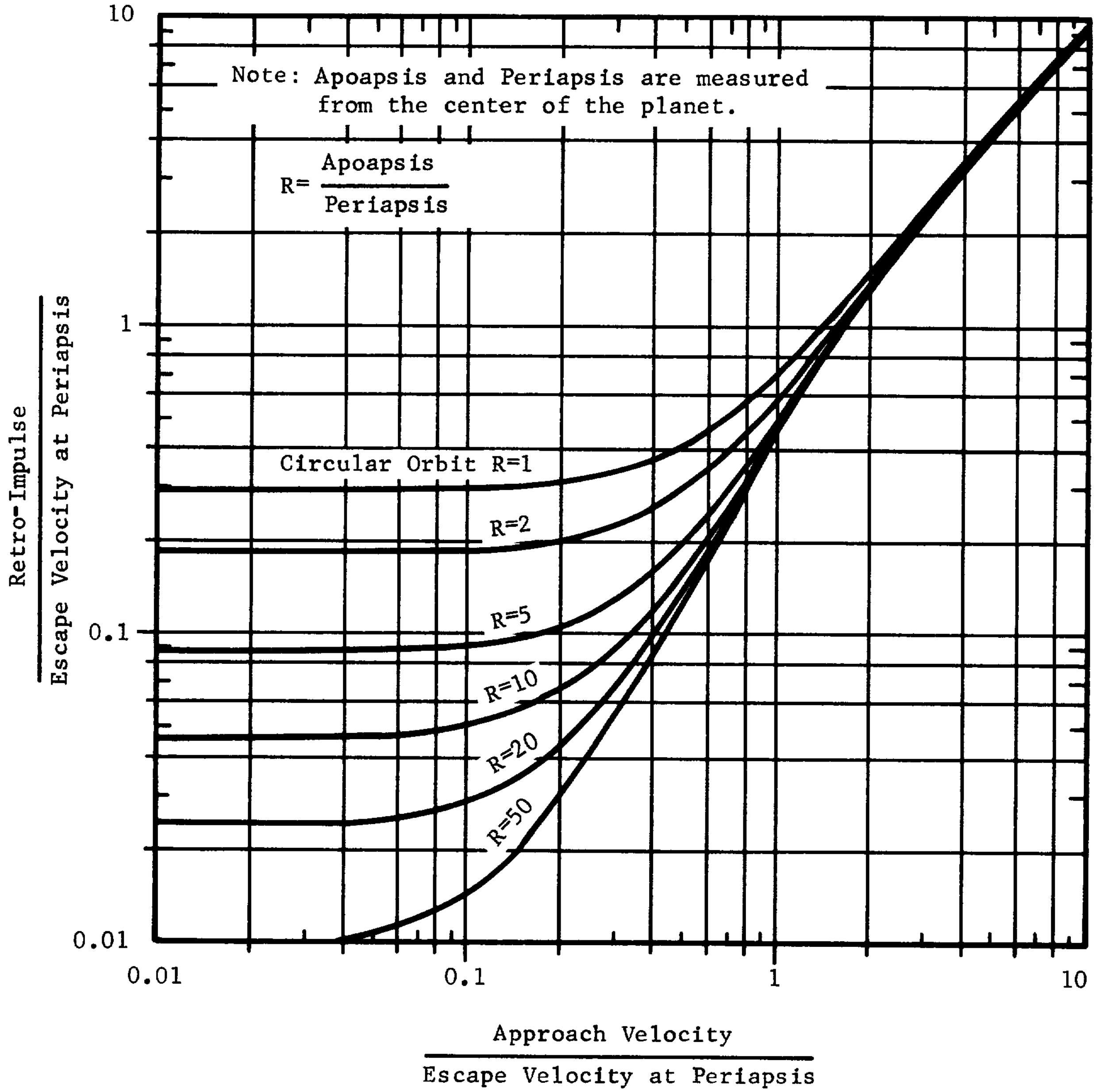


FIGURE 2-27a. RETRO-IMPULSE REQUIREMENTS VERSUS APPROACH VELOCITY FOR VARIOUS SHAPED ORBITS

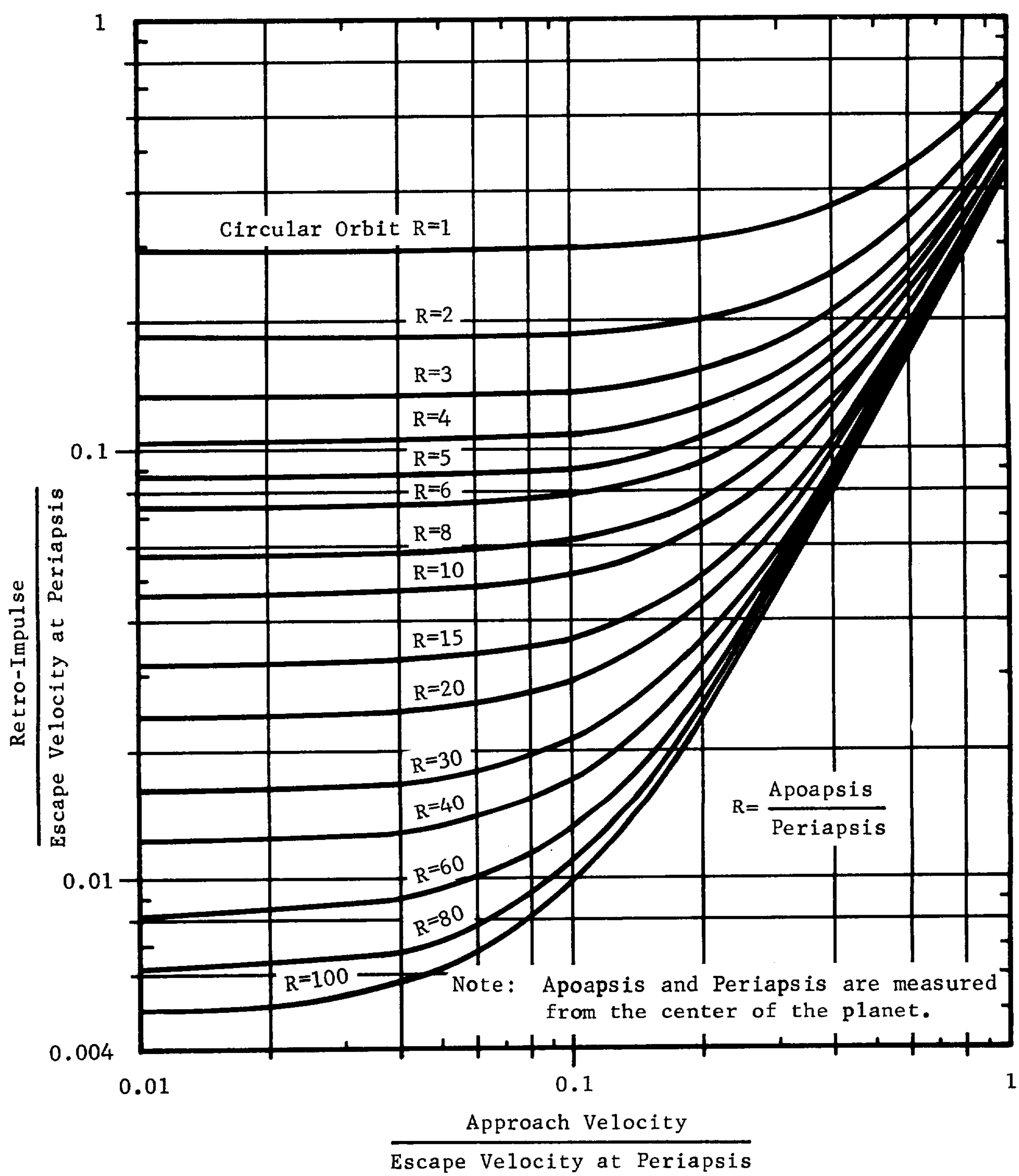


FIGURE 2-27b. RETRO-IMPULSE REQUIREMENTS VERSUS APPROACH VELOCITY FOR VARIOUS SHAPED ORBITS

(An expanded scale of Figure 2-27a)

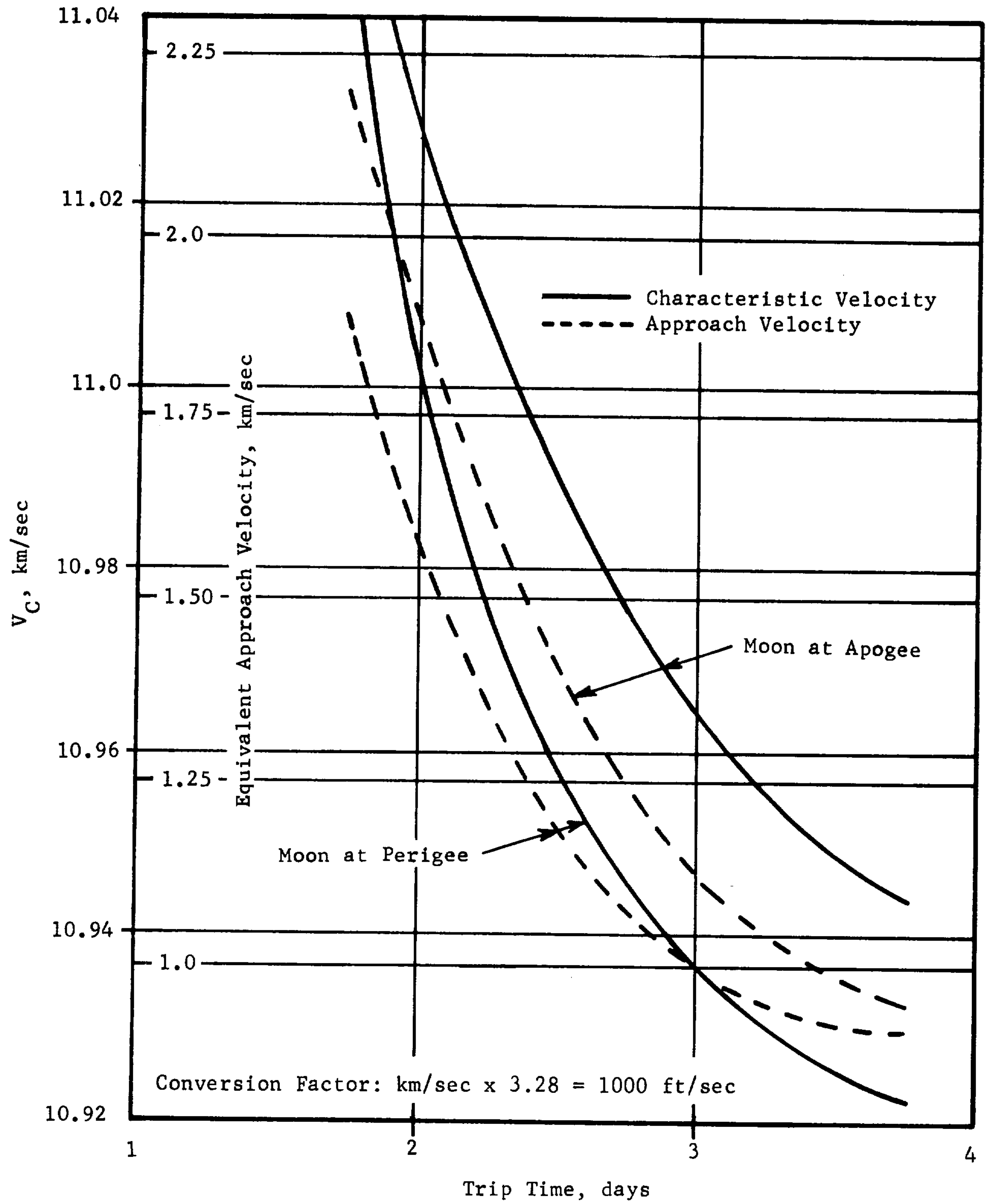


FIGURE 2-28. LAUNCH CHARACTERISTIC VELOCITY REQUIREMENTS AND APPROACH VELOCITIES FOR LUNAR MISSIONS

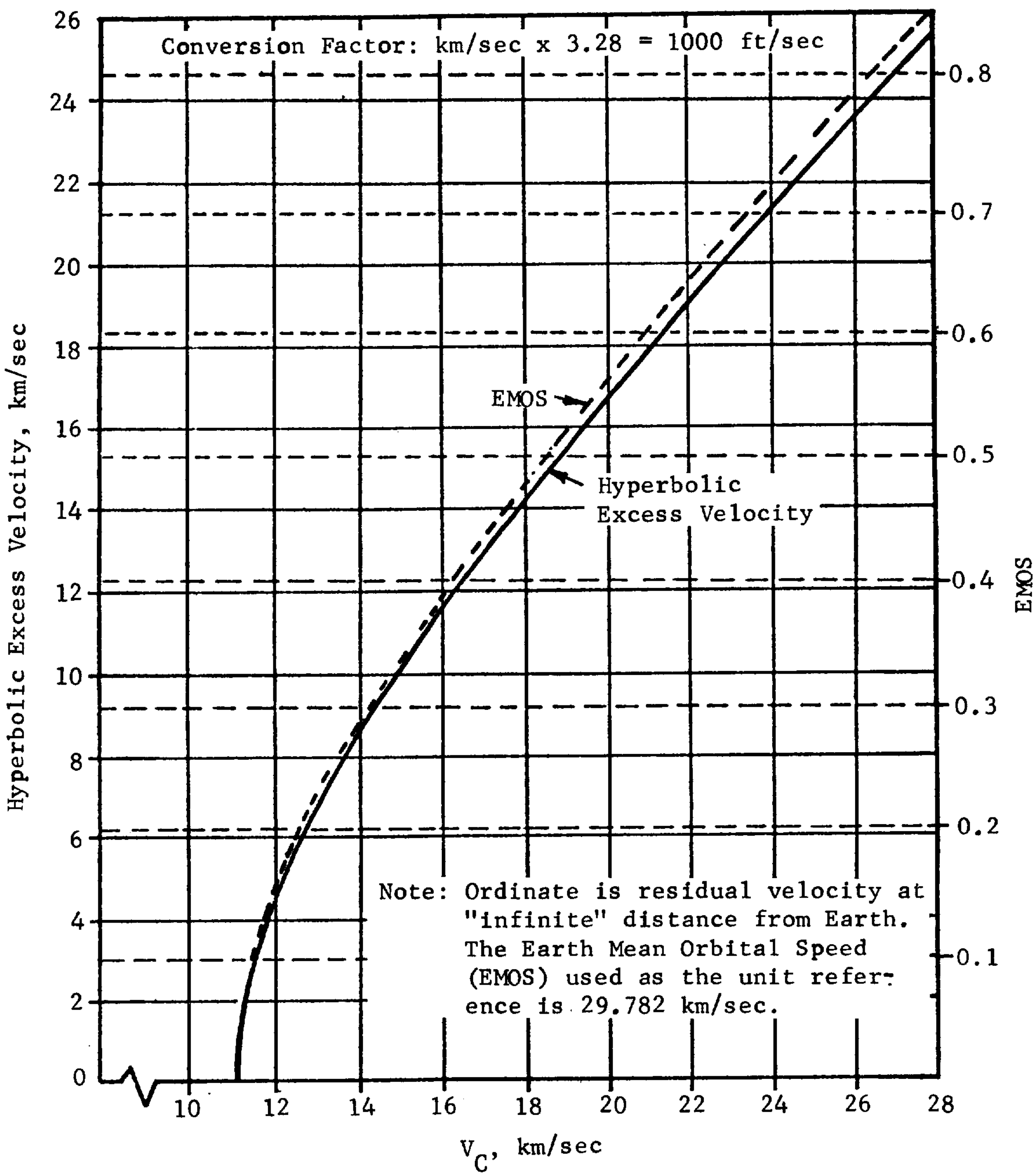


FIGURE 2-29. CONVERSION CHART, V_C TO HYPERBOLIC EXCESS VELOCITY OR EARTH MEAN ORBITAL SPEED

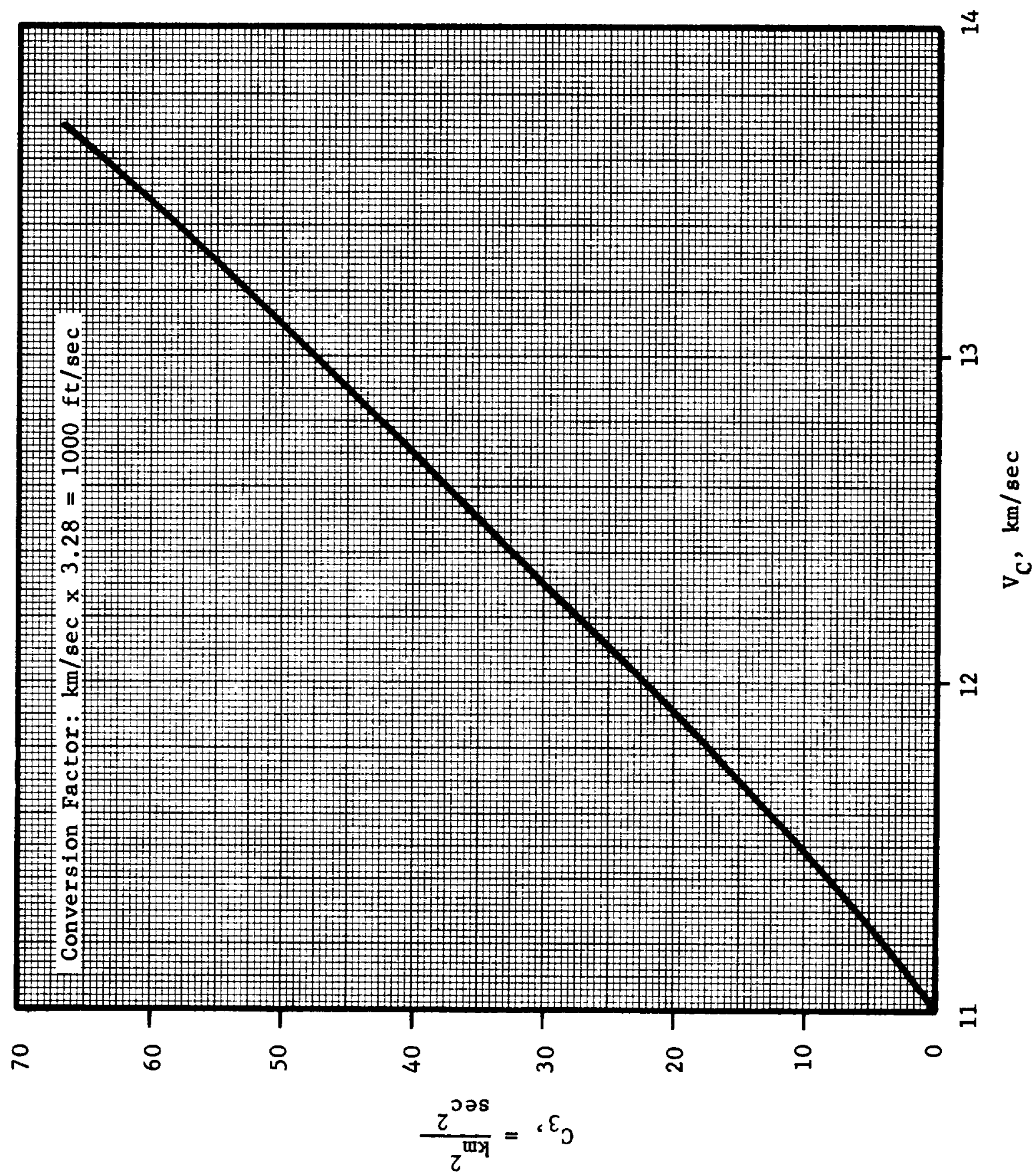


FIGURE 2-30a. CONVERSION CHART, V_C TO C_3

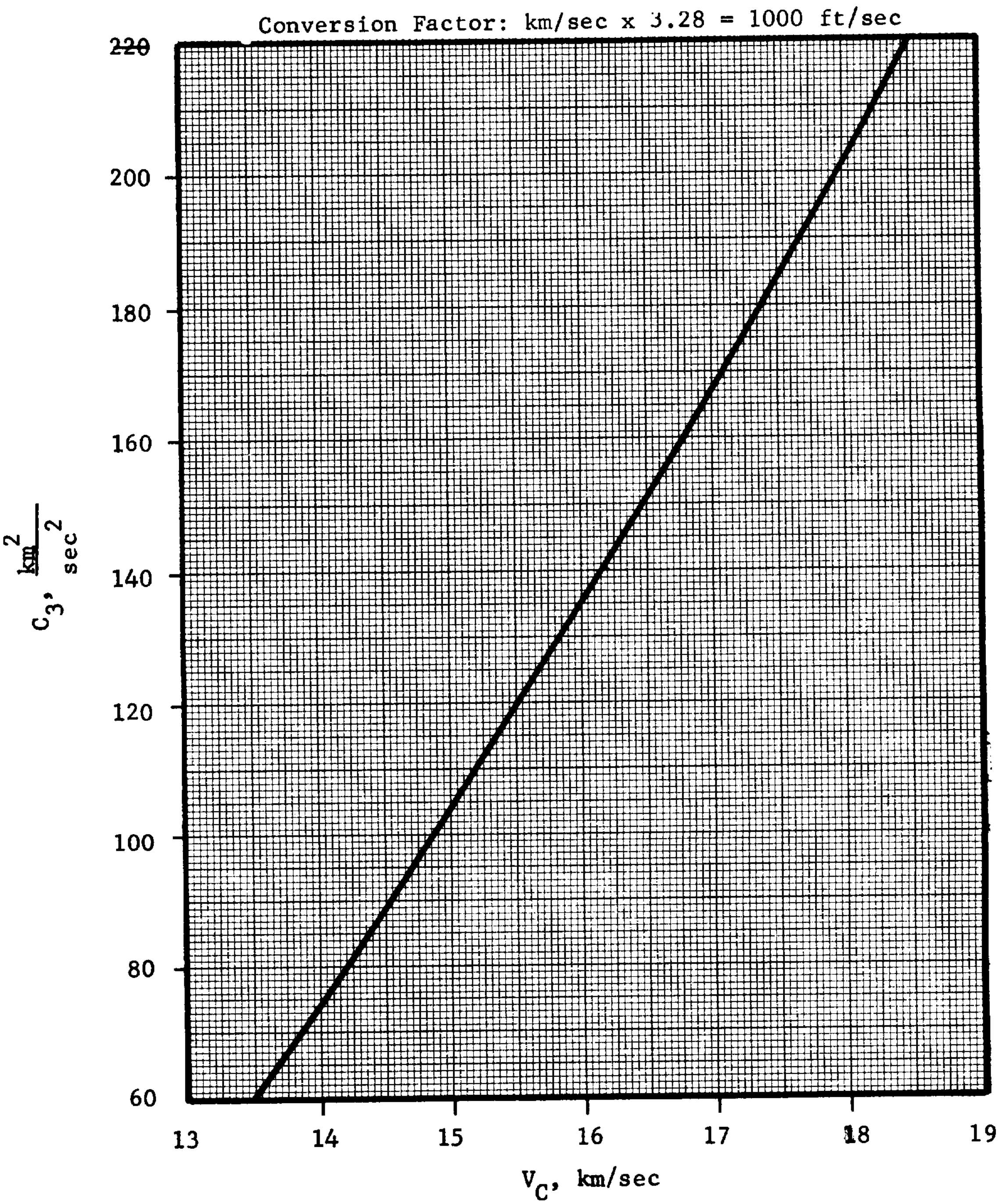


FIGURE 2-30b. CONVERSION CHART, V_C TO C_3 (Cont'd)

CHAPTER 3: EARTH ORBIT MISSION FACTORS

TABLE OF CONTENTS

CHAPTER 3: EARTH ORBIT MISSION FACTORS

300 INTRODUCTION

1. This chapter presents information for use in the planning of Earth orbital missions. Data are presented for the estimation of characteristic velocity requirements for circular and elliptical Earth orbits and for the estimation of velocity impulse requirements for orbital transfers and plane changes. These data are particularly useful for the planning of missions where energy management considerations are important.

2. A method is described for the estimation of Earth-orbit lifetimes as an aid in planning missions where this factor is significant.

301 CAUTIONARY NOTE

Data presented in this chapter are based on specific trajectories. In using these data to determine launch vehicle requirements, it is also necessary to consider the performance capabilities of particular launch vehicles such as those described in Chapter 6. Specific parking orbits and orbital maneuvers may require staging, coasting, or restarts beyond the capabilities of a particular launch vehicle. Questions concerning these matters should be referred to persons listed in the Preface.

302 EARTH ORBIT REQUIREMENTS

1. Figure 3-1 depicts the velocity required for Earth orbits. The circular orbit characteristic velocities assume a Hohmann transfer from the reference 185 km initial parking orbit. The generalized curves for circular and eccentric orbits in Figure 3-1 are not related to any particular launch site. However, the curve labeled circular equatorial orbits from ETR shows the characteristic-velocity requirements to establish a circular orbit with zero-degree inclination after launching due east from ETR. The calculation is based upon the plane change being optimally divided between the two impulses of a Hohmann transfer. Synchronous altitude is indicated on this curve.

2. More general Earth-orbital data are contained in Figure 3-2 where the total characteristic-velocity requirement for orbits of arbitrary perigee and apogee are shown. The velocity contours of Figure 3-2 are based upon an assumed two-impulse maneuver starting from a

185 km circular parking orbit. The first impulse is used to establish a transfer orbit with perigee at 185 km and apogee as shown along the abscissa. The second impulse is used to raise the perigee altitude. If sufficiently large, the second impulse could be used to establish a new apogee with that of the transfer orbit becoming the perigee of the final orbit. The more efficient of these two maneuvers is to establish the final desired apogee with the first impulse and the final desired perigee with the second impulse; however, the difference between the two techniques is significant only for very high energy orbits. The coast time between the two impulses may be constrained by system considerations. The coast time from 185 km to a transfer orbit apogee may be estimated from Figure 3-1. The velocity impulse required at the apogee of the initial transfer orbit is shown as a separate item in Figure 3-3 to assist in defining possible energy management problems or to estimate independent kick-motor requirements.

303 ORBITAL TRANSFER REQUIREMENTS

1. As mentioned previously, Figures 3-1, 3-2, and 3-3 assume an initial altitude of 185 km. This restriction does not apply to the more general method described next for estimating Earth orbital impulse requirements. This general technique must be used if the initial orbit is not at 185 km, or if a plane change is included provided that the impulses are applied at either perigee or apogee. An intermediate circular orbit may be required if the initial and final orbits do not have a common line of apsides.

2. Figures 3-4a and 3-4b illustrate the relationship between the horizontal inertial velocity at a specified reference altitude and the altitude of the other apsis for any Earth orbit. Figure 3-4a presents this relationship on a log-log scale to allow the consideration of apsis altitudes up to 1,000,000 km. Figure 3-4b is restricted to the lower altitudes and is plotted on linear coordinates with velocity increments of 100 m/sec.

3. Briefly, either Figure 3-4a or 3-4b, as appropriate, can be used to find the horizontal velocity which must exist at the reference altitude before and after an impulse is applied. The magnitude of the required impulse for coplanar orbits is then simply the absolute magnitude of the difference between these velocities. For low altitudes this difference can be obtained directly from Figure 3-4c where circular velocity has been subtracted from the orbital velocity.

4. For orbits which are not coplanar but which have a common line of apsides, Figure 3-5a or 3-5b can be used in conjunction with Figure 3-4a or 3-4b to compute the vector magnitude of the impulse required to alter the altitude of the other apsis of the initial orbit and change the plane of the orbit simultaneously. To use these figures, the initial velocity and the final velocity are obtained from Figure 3-4a or 3-4b for the reference altitude (i. e., the altitude at which the impulse is assumed to occur). As in the coplanar case, these inertial velocities are functions only of the other apsides before and after the impulse and are not dependent on the plane change. By entering Figure 3-5a or 3-5b at the computed value of the ratio of these velocities and interpolating at the specified plane change angle, the ratio of required velocity increment to final velocity can be read from the ordinate. The value obtained is the total velocity increment required to change the other apsis and the plane of the orbit simultaneously.

5. If the initial and final orbits have a common apsis altitude, a single-impulse maneuver is possible. If not, a two-impulse maneuver that may be analyzed by hypothesizing an intermediate transfer orbit is required. Usually, the intermediate transfer orbit can be formed with either apsis of the initial orbit connected to either apsis of the final orbit, but, in general, the sum of the velocity increments will be minimized if the lowest and highest altitudes possible are used for the two impulses. In particular, if a plane change is needed, it should be performed at the highest altitude possible. Although a true optimum maneuver may dictate that a small part of the total plane change should be done at the lower altitude, the improvement is usually small. In any event, regardless of the number of impulses used, a logical sequence of impulses must be specified and treated one at a time. Always use the altitude at which a particular impulse is applied as the "reference altitude" of Figure 3-4a or 3-4b. Reference 23 (Appendix B) contains example solutions based upon this technique.

304 SPECIAL EARTH ORBITS

1. Certain special types of Earth orbits are noteworthy. For example, because the Earth's oblateness causes a precession of the orbital plane about the polar axis, it is possible to select an orbit that precesses with an angular velocity equal to that of Earth about the Sun. In this case, the orbit would maintain a constant orientation with respect to a line from the Sun to Earth. Figure 3-6 presents the characteristic velocity and inclination requirements for circular Sun-synchronous orbits. The launch azimuth penalty for a WTR launch is included in the characteristic velocity. The

characteristic velocity for an ETR launch with no azimuth restrictions would differ from those shown by less than 6 m/sec.

2. For Sun-synchronous missions, the orientation of the Earth, Sun, and spacecraft are of major significance to operational success, not only from the standpoint of making observations, but also in maintaining reasonable spacecraft temperatures and solar array power. Thus, if the precession of the orbital plane due to the oblateness of the Earth differs slightly from the Earth's angular rate about the Sun, this differential drift may degrade the effectiveness of the spacecraft long before the occurrence of component failure or orbital decay due to atmospheric drag. The differential drift rate for low circular orbits (in degrees per day) as a function of orbital altitude and inclination may be determined from Figure 3-7, where a positive drift rate is eastwards. The curve with $\Delta\dot{\Omega} = 0$ (zero differential drift rate) represents Sun-synchronous orbits (as a first approximation).

3. In addition to causing a precession of the orbital plane, the oblateness of Earth also causes the line joining perigee and apogee to rotate within the orbital plane. This rotation can be eliminated if an orbital inclination of 63.4 degrees is selected. If the inclination is less than 63.4 degrees, the line of apsides rotates in the same direction as the satellite; when the inclination is greater than 63.4 degrees, the line of apsides rotates in the opposite direction.

4. Missions to the stable libration points in Earth-Moon space can be considered as special Earth orbits. These points lie about 380,000 km above the surface of Earth along lines 60 degrees on either side of the line joining Earth and the Moon. Since the influence of the Moon would be much smaller than that of Earth during establishment of this position, a good first approximation to the mission requirement assumes that the characteristic-velocity requirement is the same as that for a circular orbit with a 380,000 km altitude.

5. Additional data on Earth orbits may be found in References 24 and 25 (Appendix B).

305 EARTH-ORBIT-LIFETIME ESTIMATION

1. This section provides information to aid the mission planner in estimating orbital lifetime using initial orbit parameters, spacecraft ballistic coefficient, and launch data. For elliptical orbits, it is desirable, but not mandatory, to specify the argument of perigee, ω.

2. The most accurate orbital lifetime estimates are obtained by integrating differential equations of motion for orbiting spacecraft considering all external forces. However, this effort requires extensive input data and computer time. The accuracy of other techniques depends primarily on the assumptions made with regard to upper atmosphere density and its variation as a function of solar activity. The need to make predictions regarding solar activity during the entire orbit lifetime introduces uncertainties in the results obtained using any estimation technique. Reference 26 (Appendix B) presents a semigraphical method for the approximate prediction of orbital lifetimes based on a time-dependent atmospheric density model. The information presented in this section has been adapted from that reference and is applicable for both elliptical and circular orbits. The graphs of Figures 3-8a, 3-8b, and 3-8c give nominal lifetime factors that must then be corrected for the effects of (a) specific size, shape, and mass of the satellite (Table 3-1), (b) atmospheric density (Figure 3-10), and (c) orbit inclination and the argument of perigee, where applicable (Figure 3-9). The atmospheric density correction is based on predicted solar activities. Increased solar and geomagnetic activities shorten satellite lifetime. In mission planning, the minimum probable (or desired) lifetime is the quantity of interest. Accordingly, a reasonable upper density model is used instead of a predicted mean density model. The upper density values (Figure 3-10) were obtained by using $+2\sigma$ values for predicted solar and geomagnetic activities.

TABLE 3-1. ESTIMATION OF FREE-MOLECULAR-FLOW DRAG COEFFICIENT AND REFERENCE AREAS FOR CALCULATING BALLISTIC COEFFICIENT

Satellite Orientation	Drag Coefficient, C_d	Reference Area, S*
Stabilized body	2.06-2.2	Projected area
Simple shape	2.06	Projected area
Complex shape	2.2	Projected area
Tumbling body	2.18	1/4 total surface area

*Projected areas, S, are computed as follows:

	Nose-on (α = 0 degree)	Broadside (α = 90 degrees)
Cone	$S = (\pi D^2)/4$	$S = DL/2$
Cylinder	$S = (\pi D^2)/4$	$S = DL$

where D and L are vehicle diameter and length respectively, in meters and α = angle of attack.

3. Methodology.

a. Orbit lifetime may be estimated on the basis of the following expression used in conjunction with Table 3-1 and Figures 3-7a to 3-9:

$$\frac{L(A, P)}{365}\left(\frac{M}{C_d S}\right) f(i, \omega) = \frac{(d_e - d_\ell)^2}{Y(d_e, P) - Y(d_\ell, P)}, \qquad (3\text{-}1)$$

where

A = Initial orbit apogee altitude, km

P = Initial orbit perigee altitude, km

$L(A, P)$ = Normalized Lifetime Factor, days/kg/m^2) (see Figures 3-8a, 3-8b, and 3-8c)

$\frac{M}{C_d S}$ = Spacecraft ballistic coefficient, kg/m^2

M = Orbiting mass, kg

S = Reference area, m^2 (see Table 3-1)

C_d = Drag coefficient (see Table 3-1)

$f(i, \omega)$ = Correction factor for initial inclination and argument of perigee (see Figure 3-9)

i = Initial orbit inclination, degrees (see "Note" on Figure 3-9)

ω = Argument of perigee, degrees

d_e = Reentry date, decimal calendar years

d_ℓ = Launch date, decimal calendar years

$Y(d_e, P)$ = +2 σ solar-activity factor for reentry date (see Figure 3-10)

$Y(d_\ell, P)$ = +2 σ solar-activity factor launch date (see Figure 3-10).

b. Equation (3-1) may be solved by an iterative process after substituting appropriate values that are either specified or obtained from the indicated figures. The integrated +2 σ solar activity curves on Figure 3-10 result in conservative lifetime estimates, so that the orbit lifetime would be expected to be somewhat greater than that predicted using this procedure.

4. Sample Calculations: Three typical cases are shown to demonstrate applications of the method.

a. Elliptical Orbit

The estimating procedure is designed to give orbital lifetime when the orbital parameters, launch date and spacecraft characteristics are known. As an illustration of this procedure, assume the following parameters are known:

Launch date = January 1, 1972
Initial perigee altitude = 400 km
Initial apogee altitude = 450 km
Orbit inclination = 30 deg
Spacecraft mass = 5,000 kg
Spacecraft cross-section area = 10 m^2.

To estimate lifetime, it is necessary to solve Equation (3-1) for the quantity "$d_e - d_\ell$" (date of entry minus date of launch). First, for the given perigee and apogee altitudes, a Normalized Lifetime Factor, L(A, P), of 2.8 is found from Figure 3-8b. Second, assuming that the satellite has a simple shape and is stabilized, a drag coefficient, C_d, of 2.06 is obtained from Table 3-1 and the ballistic coefficient can be calculated as $M/C_dS = 242.5$ kg/m^2. To find the $f(i, \omega)$ correction factor, which is needed for orbits out of the equatorial plane, the argument of perigee, ω, is usually selected from values indicated on Figure 3-9; otherwise it is computed. In premission estimates, the value for ω can be computed by prescribing the latitude, φ, of the subsatellite point at perigee. Then, for a given inclination, i, the argument of perigee is $\omega = \text{Arc Sin} (\text{Sin}\,\varphi/\text{Sin}\, i)$. For $\varphi = 30°$ North and inclination $i = 30°$, $\omega = 90°$. The perigee/inclination correction factor, $f(i, \omega)$, is obtained from Figure 3-9 as 0.93. The preceding values can now be substituted into Equation 3-1, which reduces to:

$$\frac{(d_e - d_\ell)^2}{Y(d_e, P) - Y(d_\ell, P)} = \frac{2.8}{365} \times (242.5) \times (0.93) = 1.730 \quad (3\text{-}2)$$

Next, using Figure 3-10, a lifetime must be found which gives the proper combination of +2 σ Solar Activity Factors satisfying Equation (3-3). Preliminary sample iterations using Equation (3-3) and Figure 3-10 indicate that the lifetime will be somewhere between 5 and 6 years. Successive iterations show that the proper combination of values from Figure 3-10 which simultaneously satisfy Equation (3-3) is:

$$\frac{(1977.14-1972.0)^2}{Y(d_e, P) - Y(d_\ell, P)} = \frac{(5.14)^2}{(17.5 - 2.2)} = 1.730 \quad , \tag{3-3}$$

so that the predicted orbit lifetime is 5.14 years.

b. Circular Orbits

In this case, the apogee and perigee altitudes are equal and are used to enter one of the appropriate graphs (Figures 3-8a, 3-8b, 3-8c) for L(A, P). The recommended f(i, ω) factor is the lowest for the desired inclination; this provides the most conservative lifetime estimate. All other computations are made as discussed in subparagraph 305.4a.

c. Inverse Case

An inverse procedure may be used to estimate minimum orbit altitudes when a desired lifetime is specified. Typically, the launch date, spacecraft characteristics, and orbit inclination are given or can be assumed:

Launch date = January 1, 1976
Orbit inclination = 90 degrees
Spacecraft ballistic coefficient = 242.5 kg/m^2
(same as previous example)
Desired lifetime = 4 years (reentry date, January 1, 1980)

The inverse of the procedure illustrated in subparagraph 305.4a begins by estimating an initial perigee altitude, say 400 km, and obtaining the +2 σ Solar Activity Factors from Figure 3-10 for the launch and entry dates. The right side of Equation (3-1) can now be evaluated:

$$\frac{L(A, P)}{365} \frac{M}{C_d S} f(i, \omega) = \frac{4^2}{19.5 - 15.9} = 4.45 \quad . \tag{3-4}$$

The ballistic coefficient is given (242.5 kg/m^2) and a Perigee/Inclination Correction Factor can be found upon specifying the desired perigee latitude. Assuming a φ = 45 degrees South for this example, the argument of perigee is found to be ω = -45 or +315 degrees. Thus, a value of 1.0 is obtained from Figure 3-9 for the Perigee/Inclination Correction Factor. The appropriate values can now be substituted into Equation (3-4) which can be solved for the Normalized Lifetime Coefficient:

$$L(A, P) = \frac{4.45 \times 365}{242.5 \times 1.0} = 6.70 \quad (3\text{-}5)$$

Entering Figure 3-7c with this value, it is found that an apogee of 600 km gives the desired 4-year lifetime for the assumed perigee of 400 km. If a near-circular orbit is desired, a higher perigee can be assumed and the estimation process can be repeated. In this case, a near-circular orbit of about 480 km altitude gives the desired 4-year lifetime.

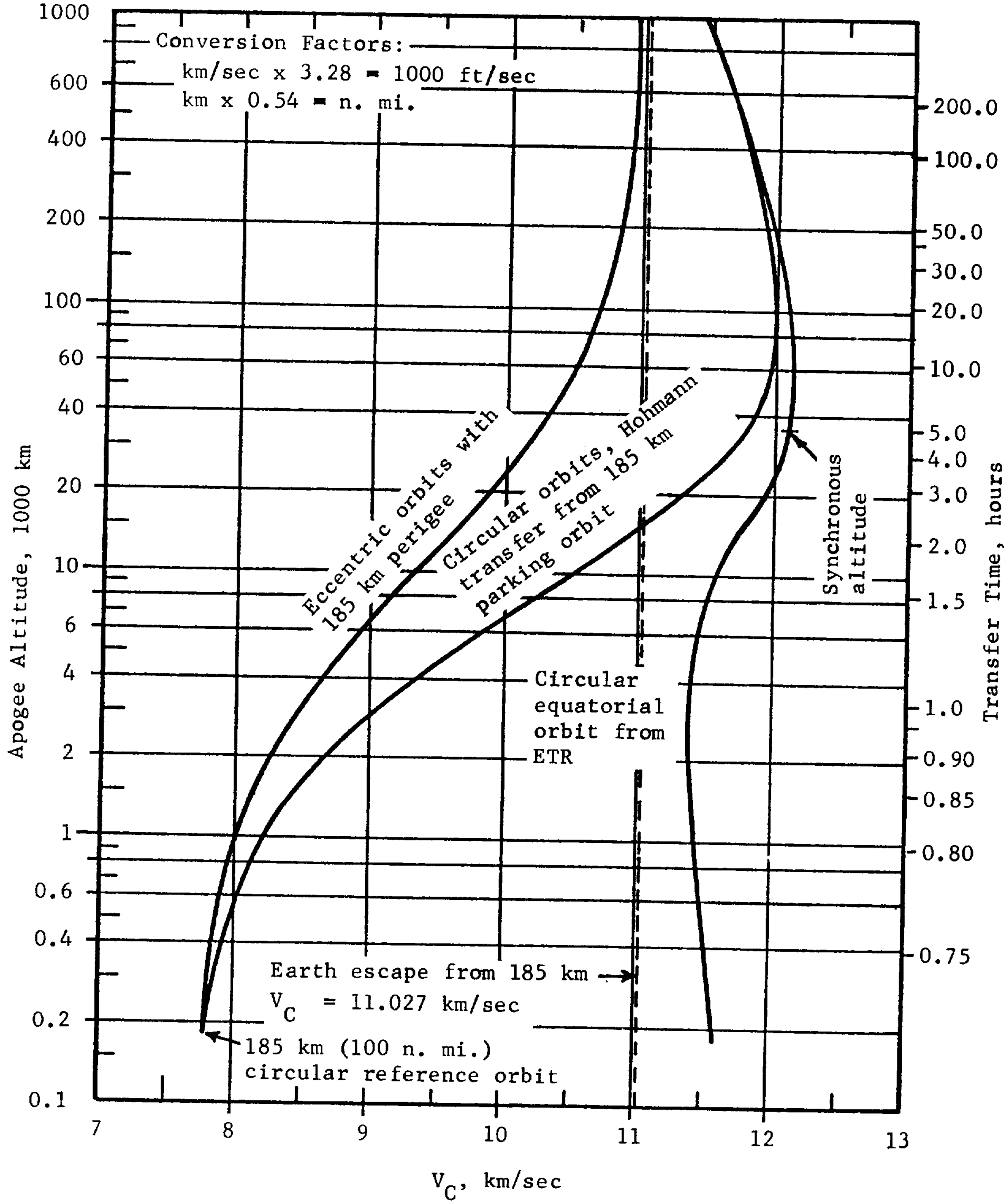


FIGURE 3-1. VELOCITY REQUIRED FOR EARTH ORBITS

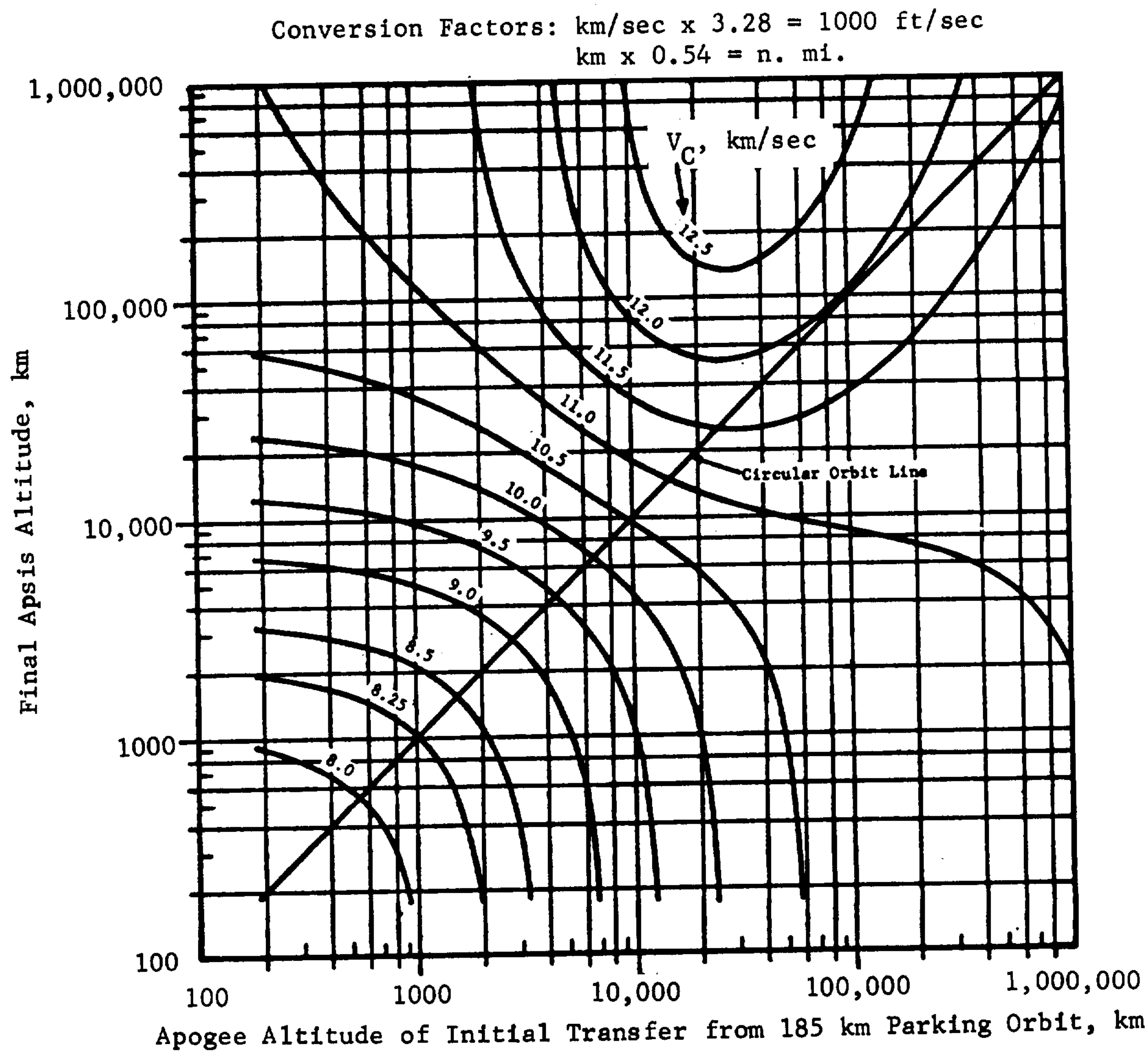


FIGURE 3-2. TOTAL V_C REQUIREMENTS ASSUMING TWO-IMPULSE TRANSFER FROM 185 KM ORBIT

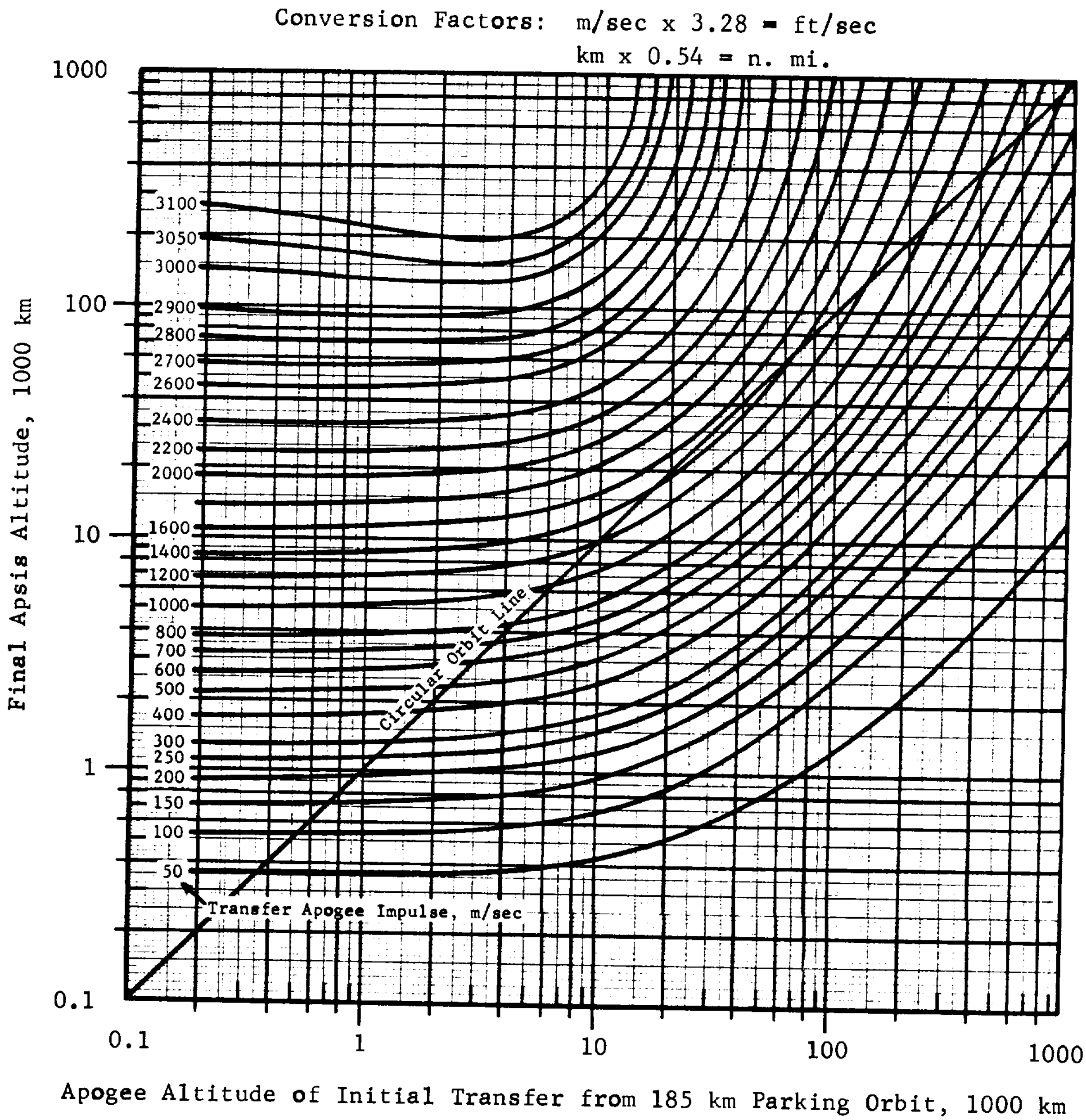


FIGURE 3-3. TRANSFER APOGEE IMPULSE REQUIREMENTS AFTER TRANSFER FROM 185 KM ORBIT

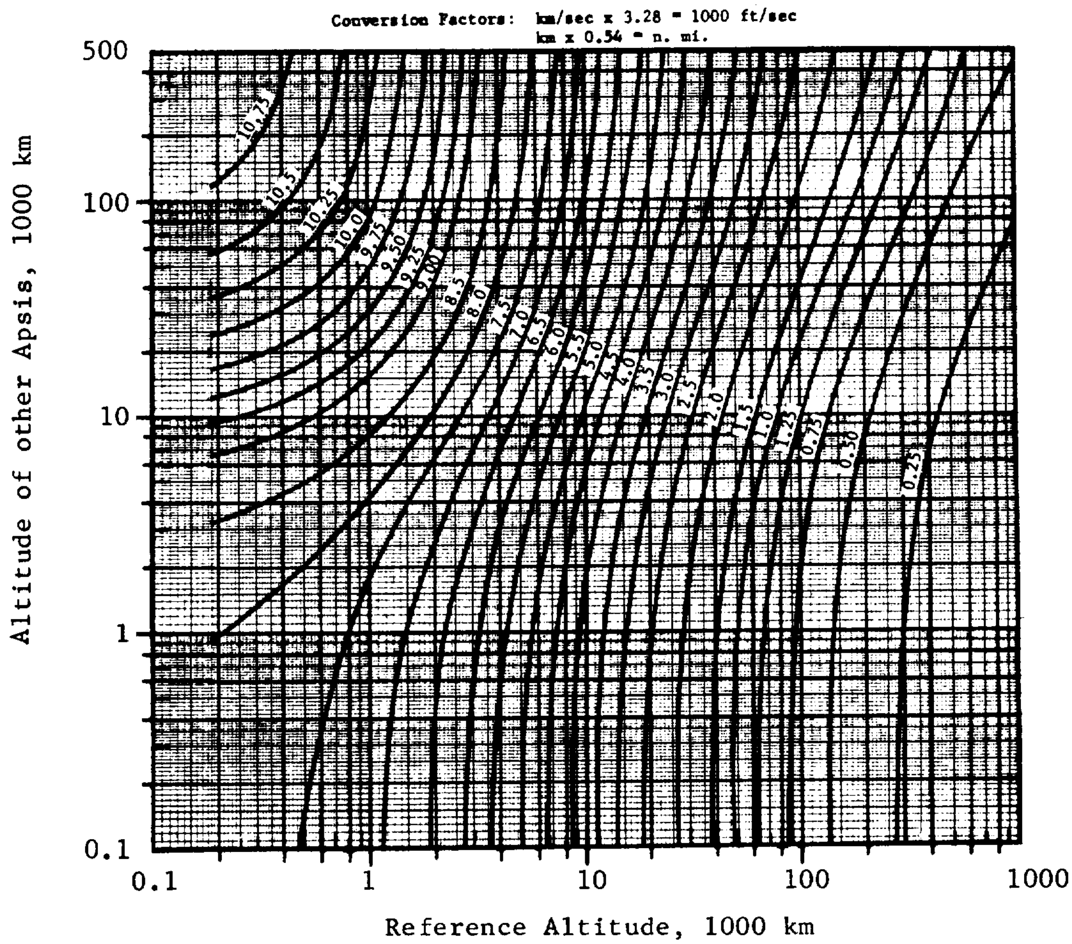


FIGURE 3-4a. HORIZONTAL VELOCITY REQUIRED AT REFERENCE ALTITUDE TO ESTABLISH OTHER APSIS

Note: Numbers on curves indicate velocity in km/sec.

Conversion Factors: km/sec x 3.28 = 1000 ft/sec
km x 0.54 = n. mi.

Note: Numbers on curves indicate V_C, km/sec

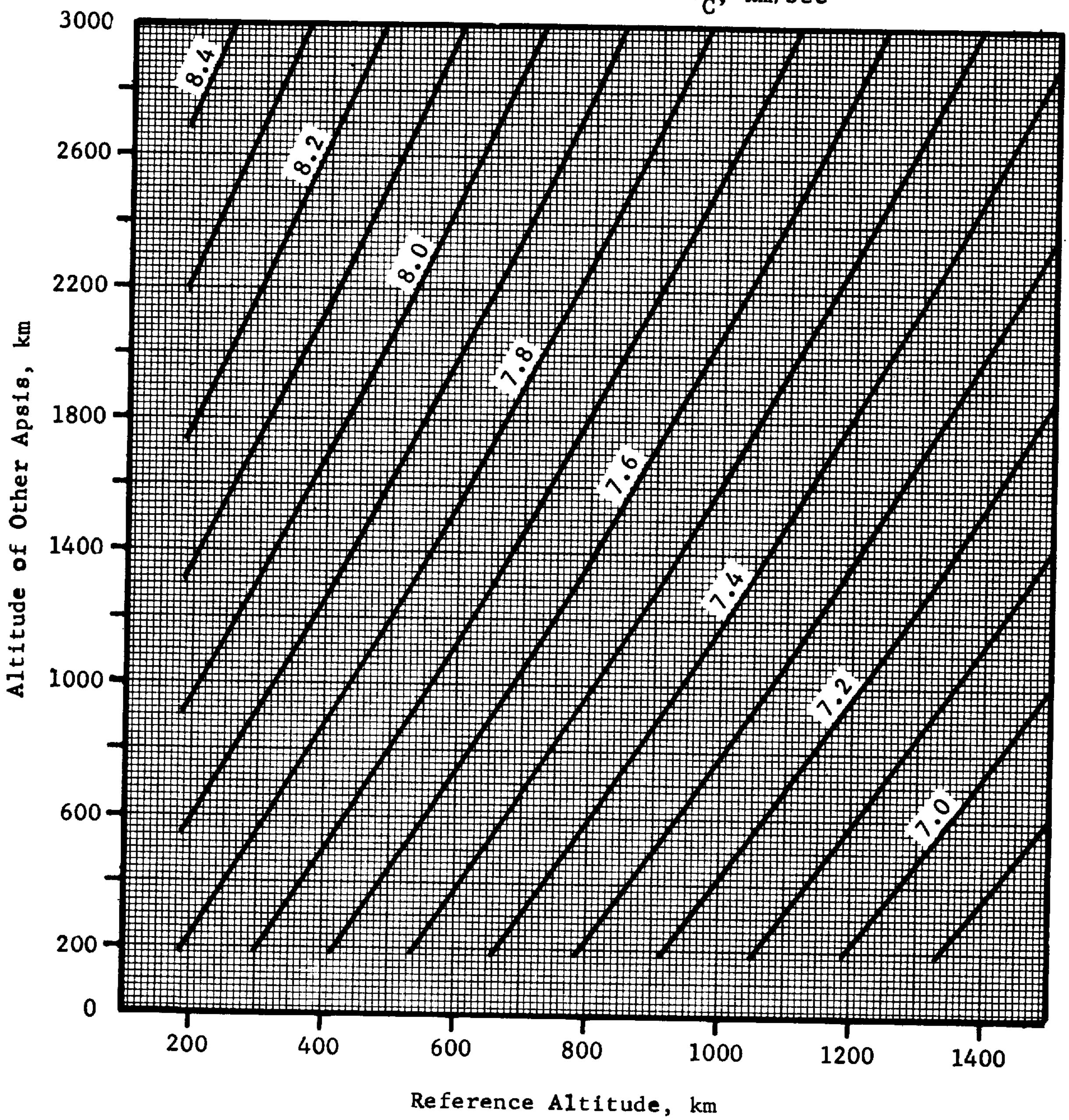


FIGURE 3-4b. HORIZONTAL VELOCITY REQUIRED AT REFERENCE ALTITUDE TO ESTABLISH OTHER APSIS (LOWER ALTITUDES)

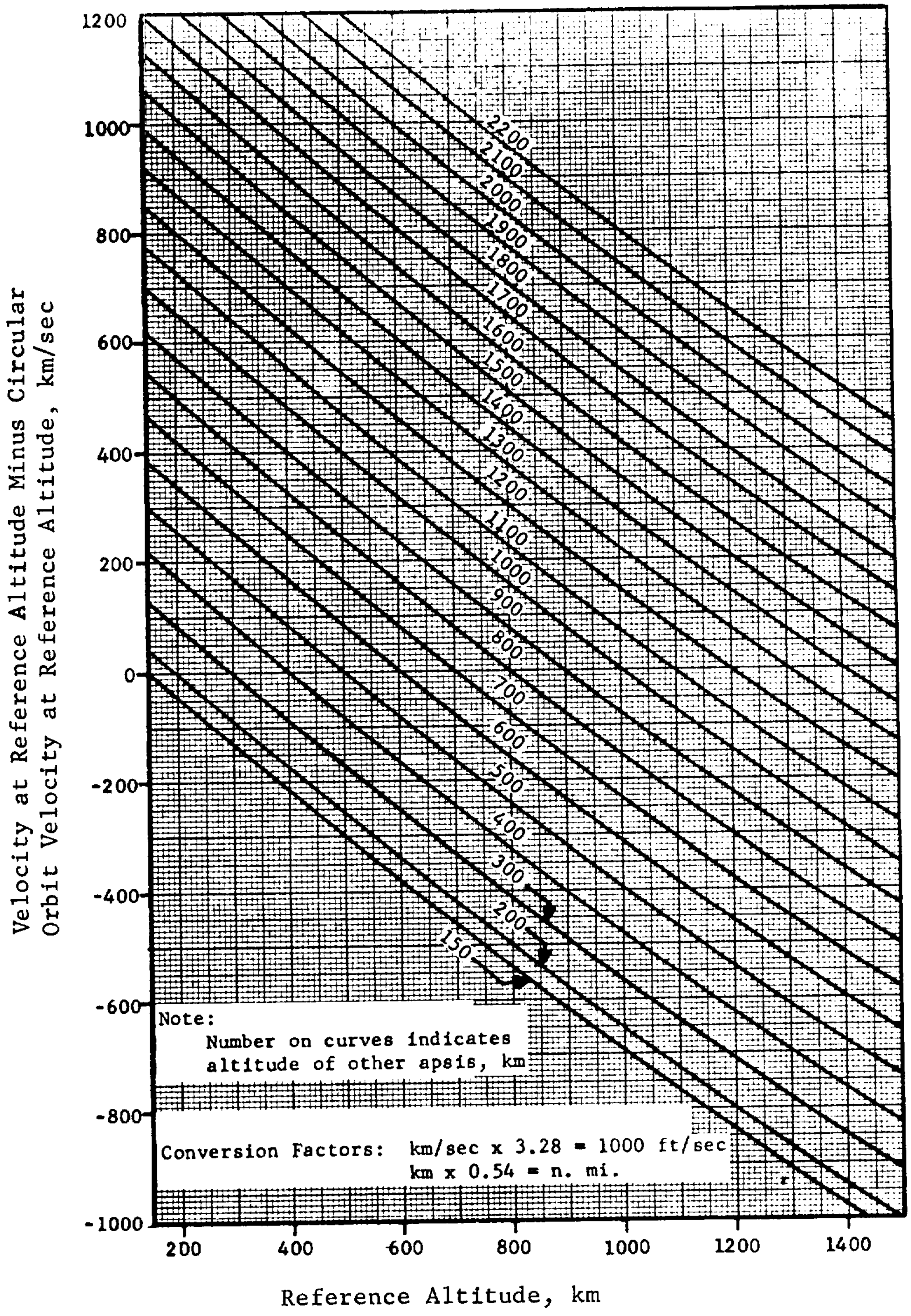


FIGURE 3-4c. HORIZONTAL VELOCITY IMPULSE AT REFERENCE ALTITUDE REQUIRED TO ESTABLISH OTHER APSIS

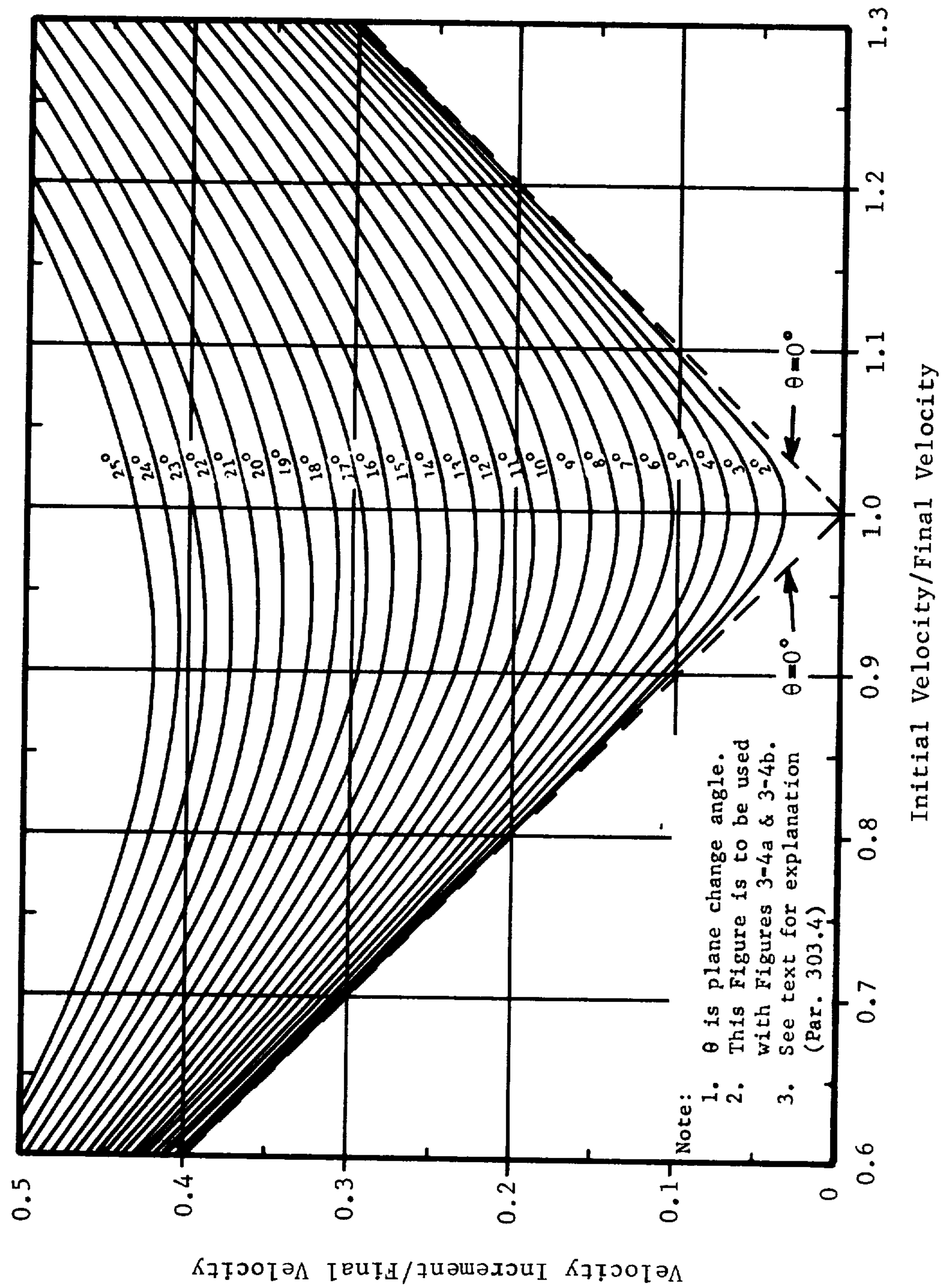


FIGURE 3-5a. NORMALIZED VELOCITY INCREMENT FOR NONCOPLANAR ORBIT TRANSFER

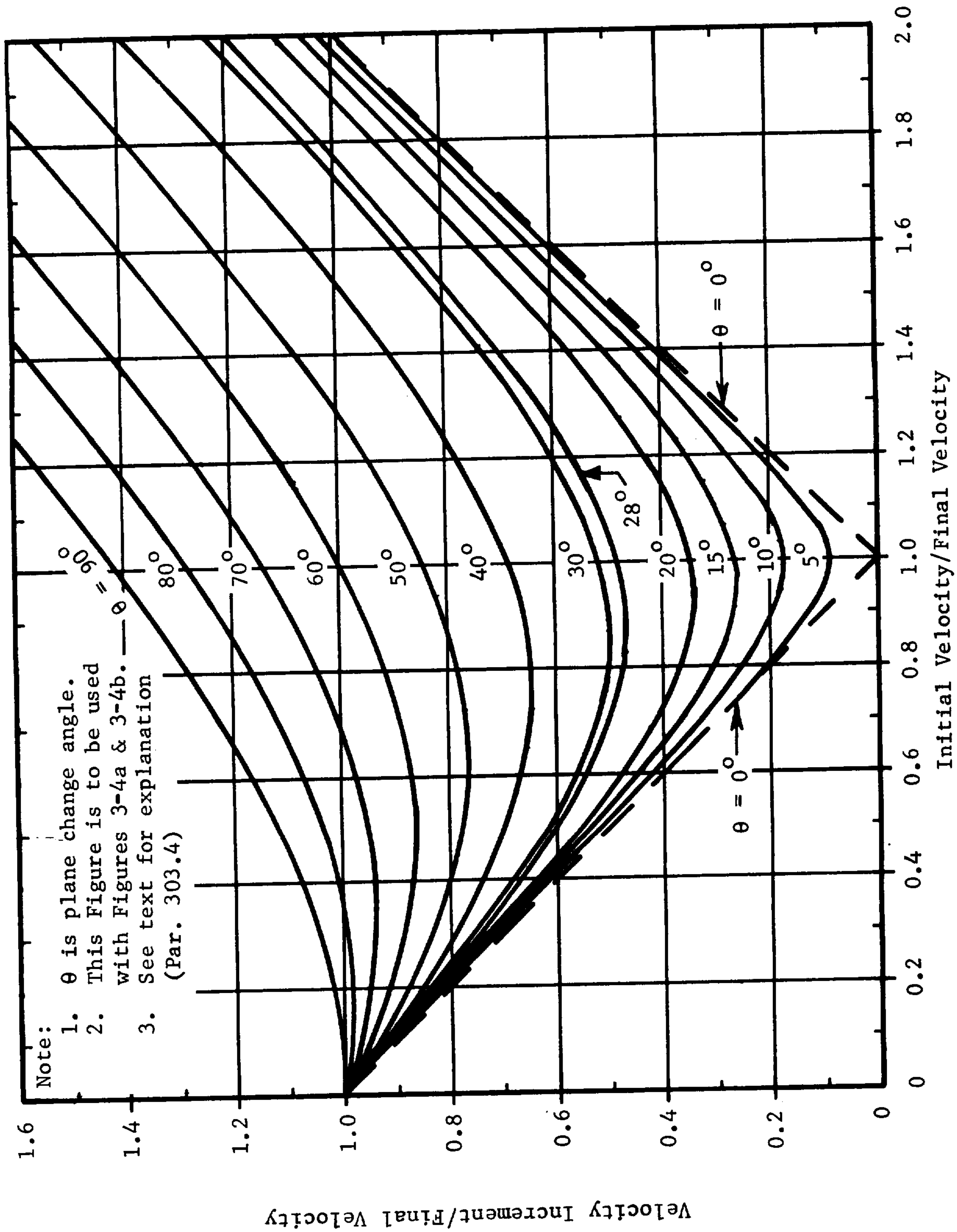


FIGURE 3-5b. NORMALIZED VELOCITY INCREMENT FOR NONCOPLANAR ORBIT TRANSFERS (EXTENDED SCALE)

Conversion Factors: km/sec x 3.28 = 1000 ft/sec
km x 0.540 = n. mi.

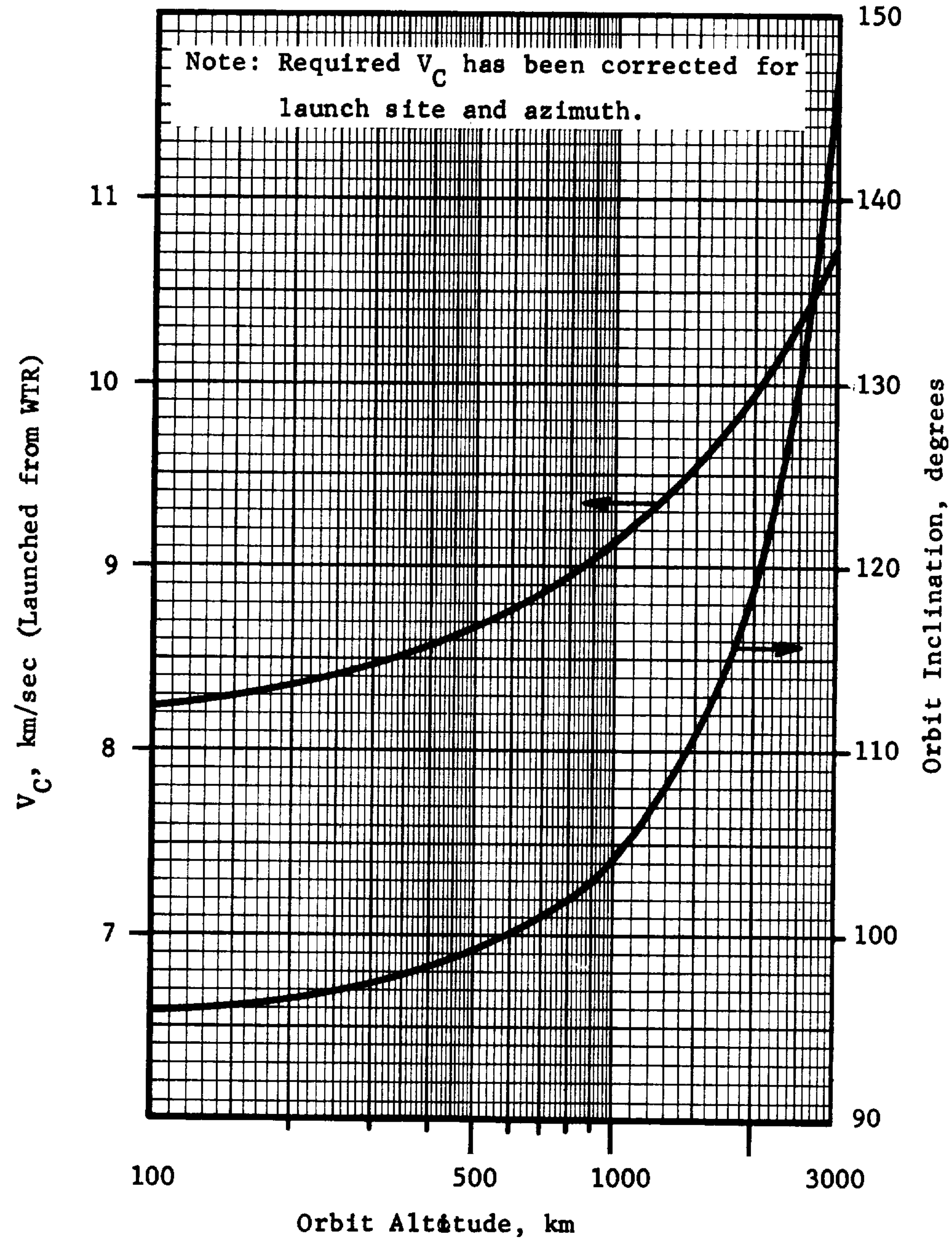


FIGURE 3-6. CHARACTERISTICS OF CIRCULAR SUN-SYNCHRONOUS ORBITS

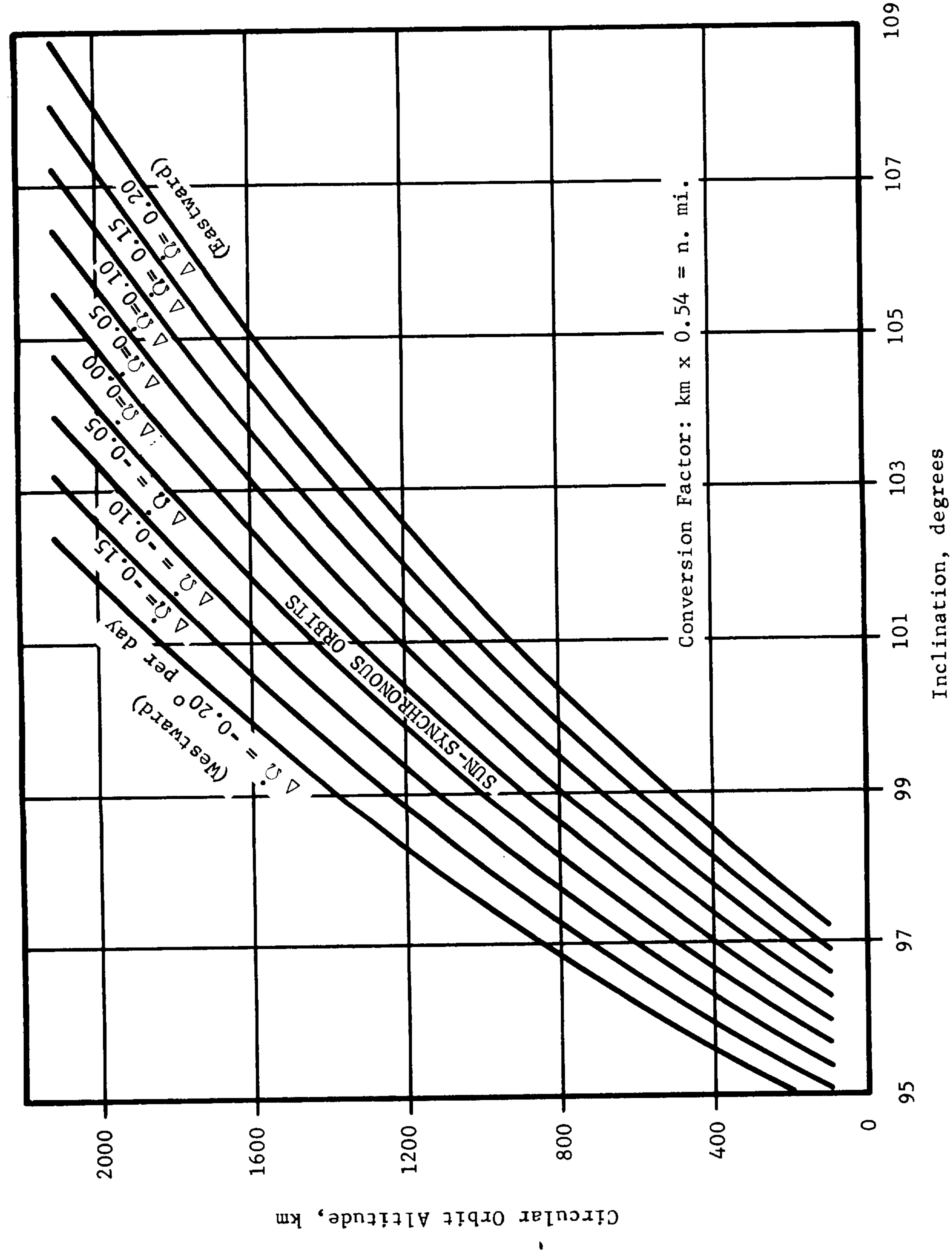


FIGURE 3-7. DIFFERENTIAL DRIFT RATES FOR LOW SUN-SYNCHRONOUS ORBITS

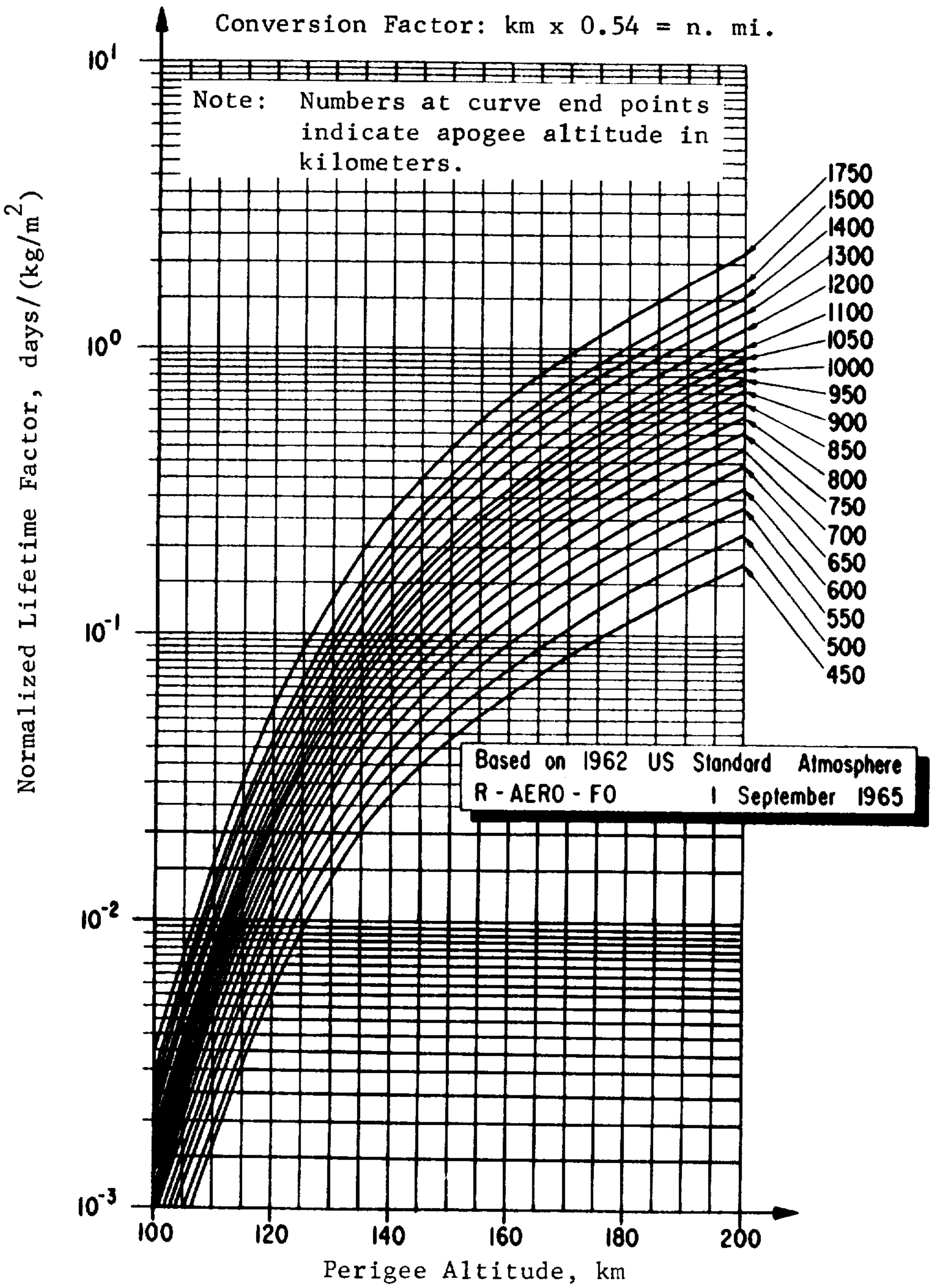


FIGURE 3-8a. EARTH ORBITAL LIFETIME FACTOR

(Perigee: 100 - 200 km
Apogee: 450 - 1750 km)

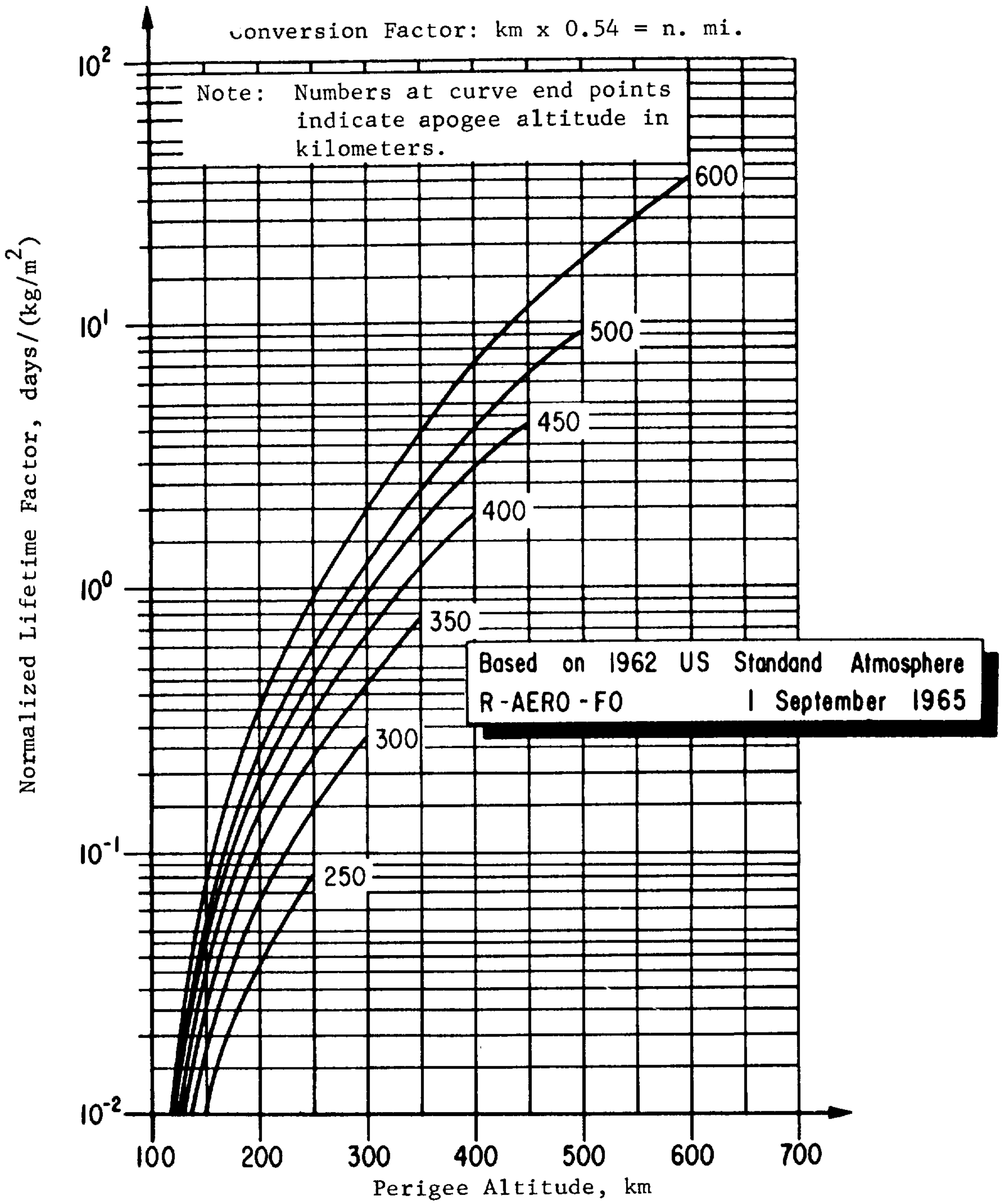


FIGURE 3-8b. EARTH ORBITAL LIFETIME FACTOR

(Perigee: 100 - 600 km
Apogee: 250 - 600 km)

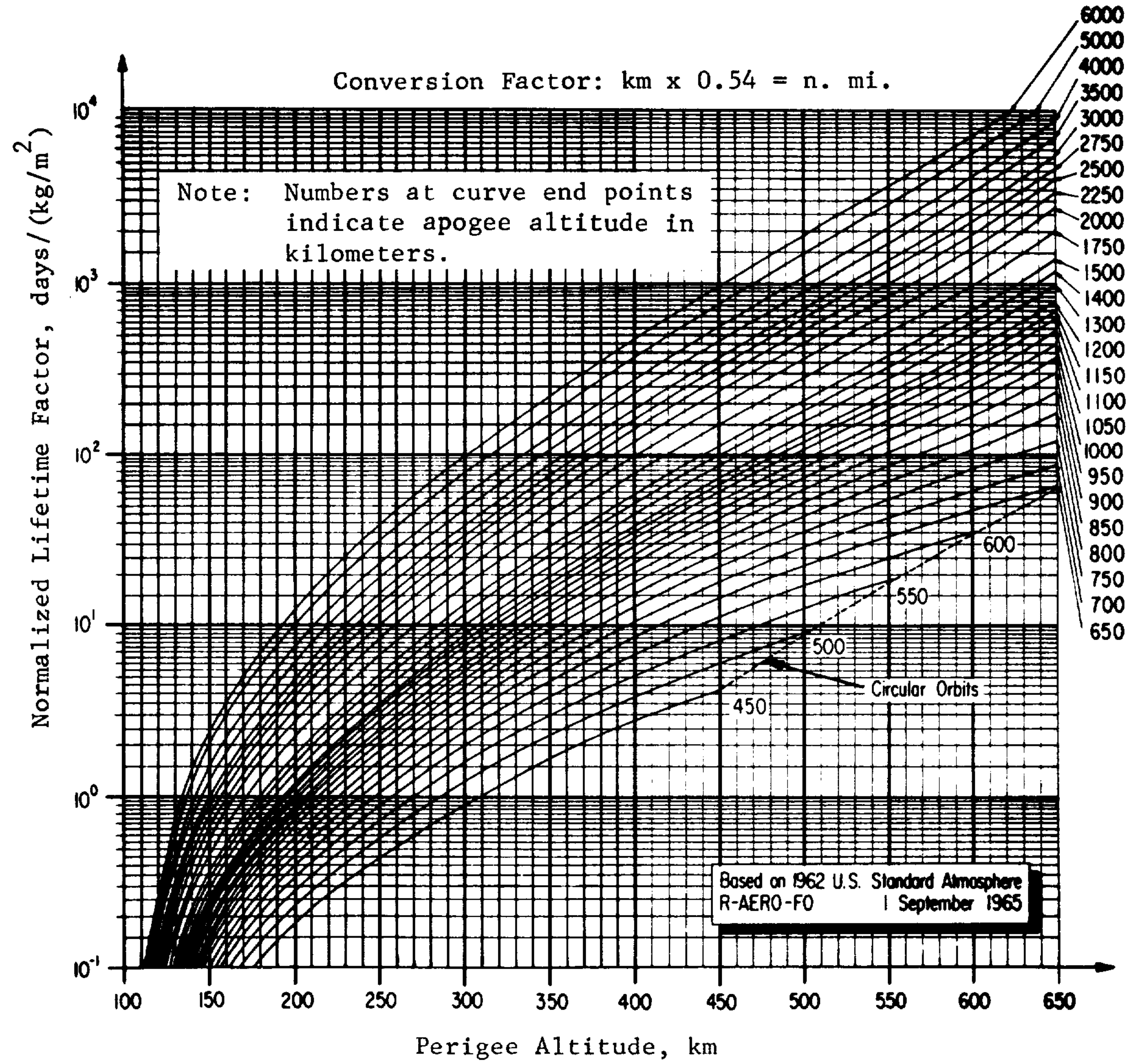
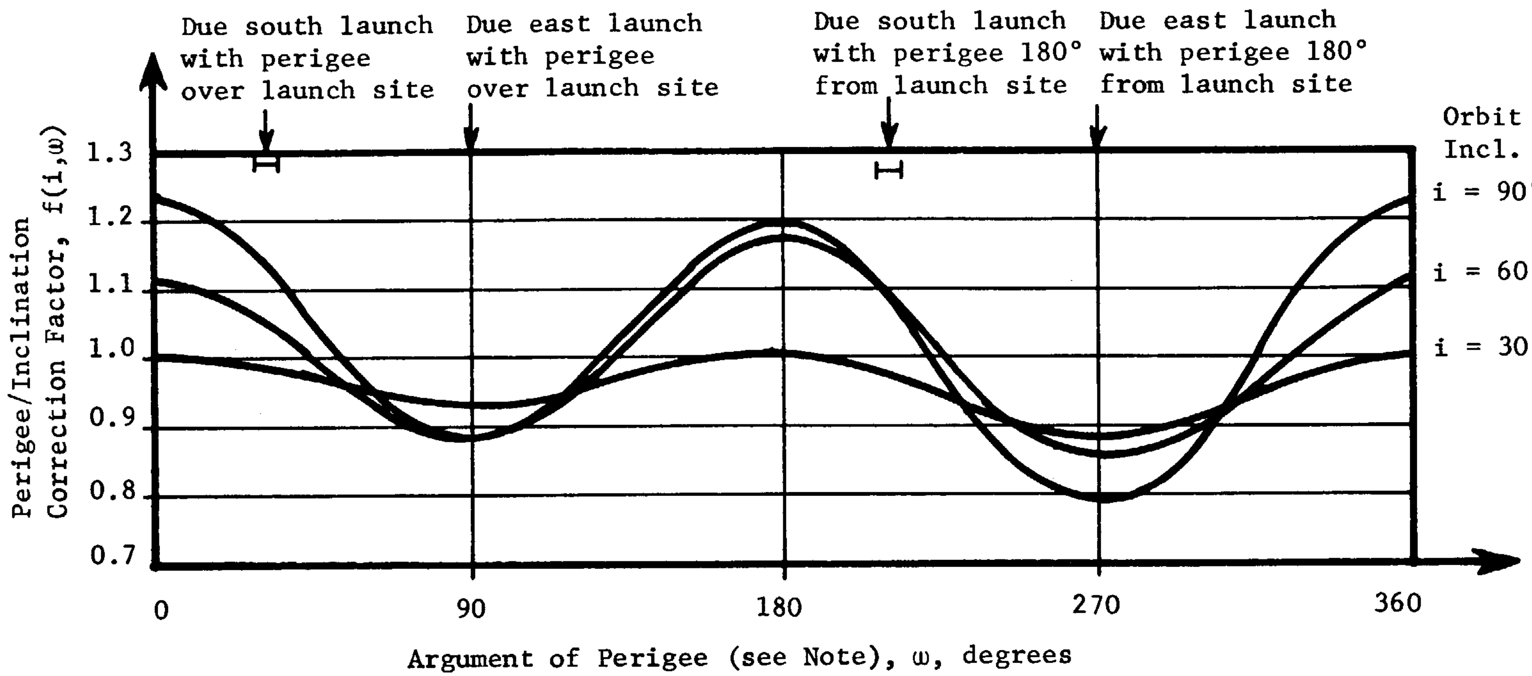


FIGURE 3-8c. EARTH ORBITAL LIFETIME FACTOR

(Perigee: 100 - 650 km
Apogee: 450 - 6000 km)



Note: Argument of perigee, ω, is the angle measured from the line of ascending nodes to the perigee radius. It can be determined from the expression:

$$\sin \omega = \pm \frac{\sin \varphi_p}{\sin i} \begin{cases} + \text{ when } \varphi_p = \text{N Lat} \\ - \text{ when } \varphi_p = \text{S Lat} \end{cases}$$

where φ_p is the latitude of perigee and i is the orbit inclination.

While ω may assume any value between 0 and 360 degrees, several typical values are indicated on the Figure.

FIGURE 3-9. PERIGEE/INCLINATION CORRECTION FACTOR FOR ORBIT LIFETIME ESTIMATIONS

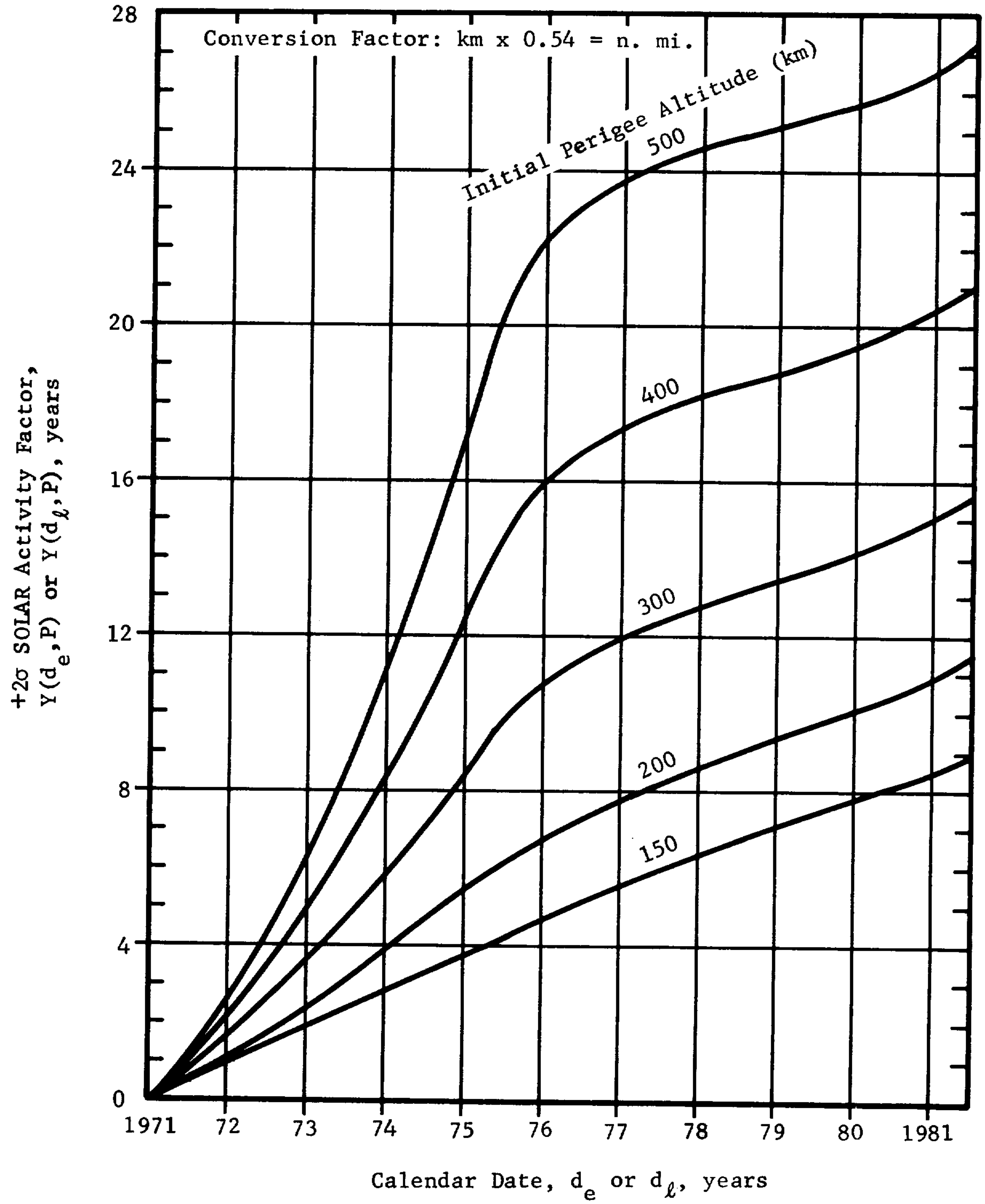


FIGURE 3-10. +2 σ SOLAR ACTIVITY FACTOR FOR ORBIT LIFETIME ESTIMATIONS

CHAPTER 4. LAUNCH-SITE FACTORS

TABLE OF CONTENTS

CHAPTER 4: LAUNCH-SITE FACTORS

400 LAUNCH SITES

1. The Eastern Test Range (ETR) is used for launches for which it is feasible to employ the rotation of Earth to increase the velocity of the vehicle – that is, for launches predominantly eastward. The Western Test Range (WTR) is used chiefly for southerly launches, often slightly retrograde, for near-polar orbits. Scout is the only launch vehicle considered in this document for which there are facilities at Wallops Island. Scout is also launched by Italy from the San Marco Platform near the equator on the east coast of Africa.

2. The vehicles for which launch facilities are available or planned at the Eastern and Western Test Ranges, and at San Marco and Wallops Island are as follows:

Eastern Test Range	Western Test Range	Wallops Island and San Marco
Delta (Various Configurations)	Scout	Scout
SLV3D/Centaur	Thor/Burner II	
Titan IIIC	Delta (Various Configurations)	
Titan IIID	SLV3A/Burner II	
Titan IIIE/Centaur	TAT(3C)/Agena	
Saturn IB	Titan IIIB/Agena	
Saturn V	Titan IIID	

401 VELOCITY PENALTIES

1. The figures in this chapter give an approximate velocity increment that must be either added to the mission velocity requirement or subtracted from the launch vehicle capability for any launch that is not in an eastward direction from the ETR.

Figure 4-1 shows the velocity penalty and launch azimuth as functions of orbit inclination for the ETR. Figures 4-2, 4-3, and 4-4 present data similar to those in Figure 4-1 for the WTR, Wallops Island, and San Marco, respectively. Azimuth angle is measured in a clockwise direction from geographical north in the horizontal plane at the launch point. The inclination of an Earth orbit is defined as the angle between the angular momentum vector of the orbit and the North Pole. This is equivalent to the angle between the Earth's equatorial plane and the plane of the orbit for prograde orbits and 180° minus this angle for retrograde orbits. The corrections shown are approximated as the difference between the local Earth surface velocity in the direction of launch and the Earth surface velocity in an eastward direction at the ETR. A more precise determination of these corrections would require consideration of other factors related to the launch.

2. Figures 4-1, 4-2, 4-3, and 4-4 also display generalized limits imposed on launch azimuth because of range safety considerations. There are specific range safety limits associated with each launch vehicle. These safety limits can be waived, but flights outside these limits require special clearances. Questions on the subject should be referred to persons listed in the Preface.

3. The launch azimuth corresponding to a given orbit inclination was computed by assuming that injection into a 185 km orbit takes place directly over the launch site. The relationships between launch azimuth and orbit inclination, for the four launch sites, are shown for a broader range of inclinations in Figure 4-5. For orbital inclinations unattainable by direct injection, a plane change maneuver must be performed. For final orbits other than circular, specific calculations must be performed considering apses and angles involved. Refer to pars. 302 and 303 for the appropriate procedure.

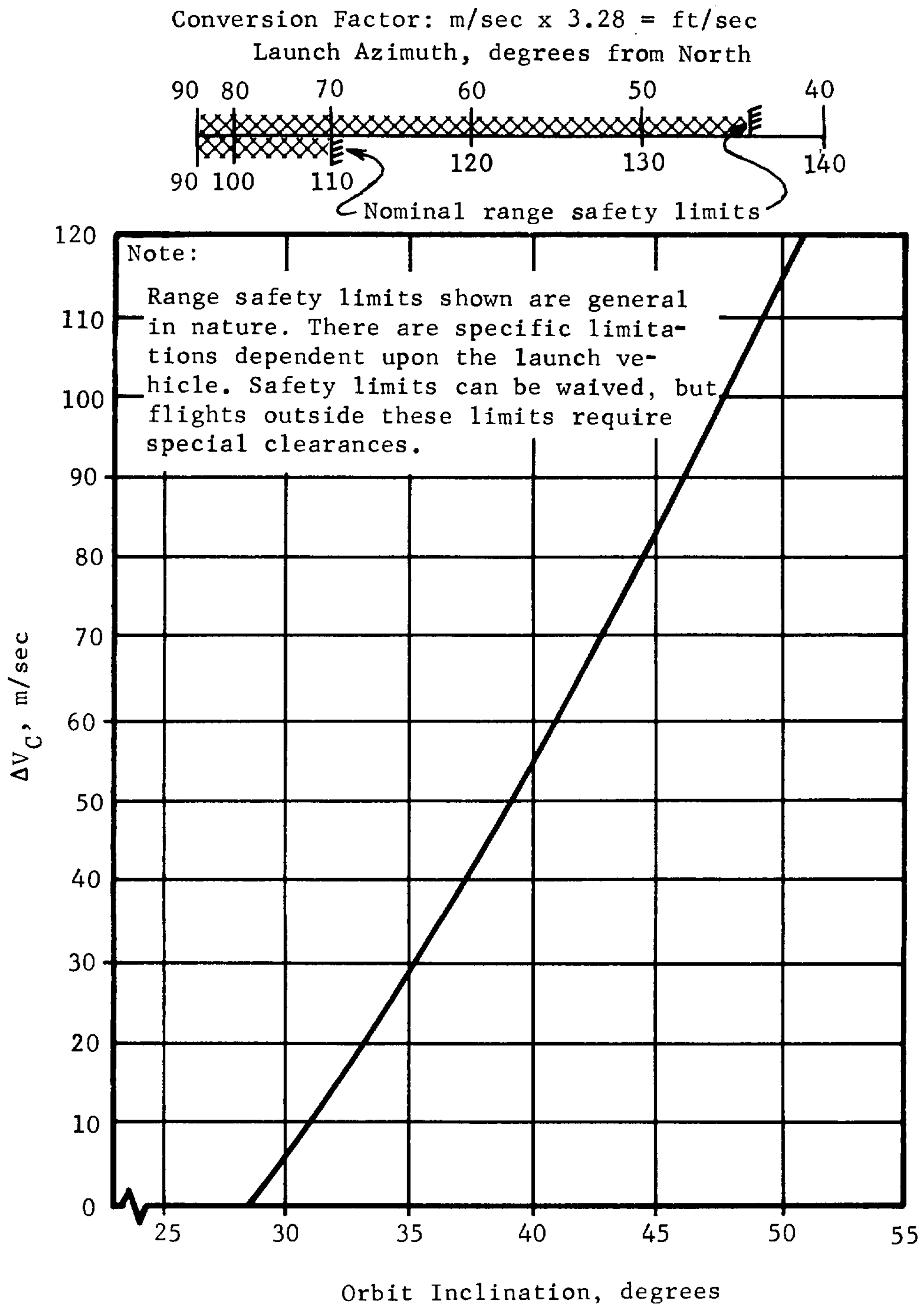


FIGURE 4-1. VELOCITY PENALTY FOR LAUNCHES FROM THE EASTERN TEST RANGE

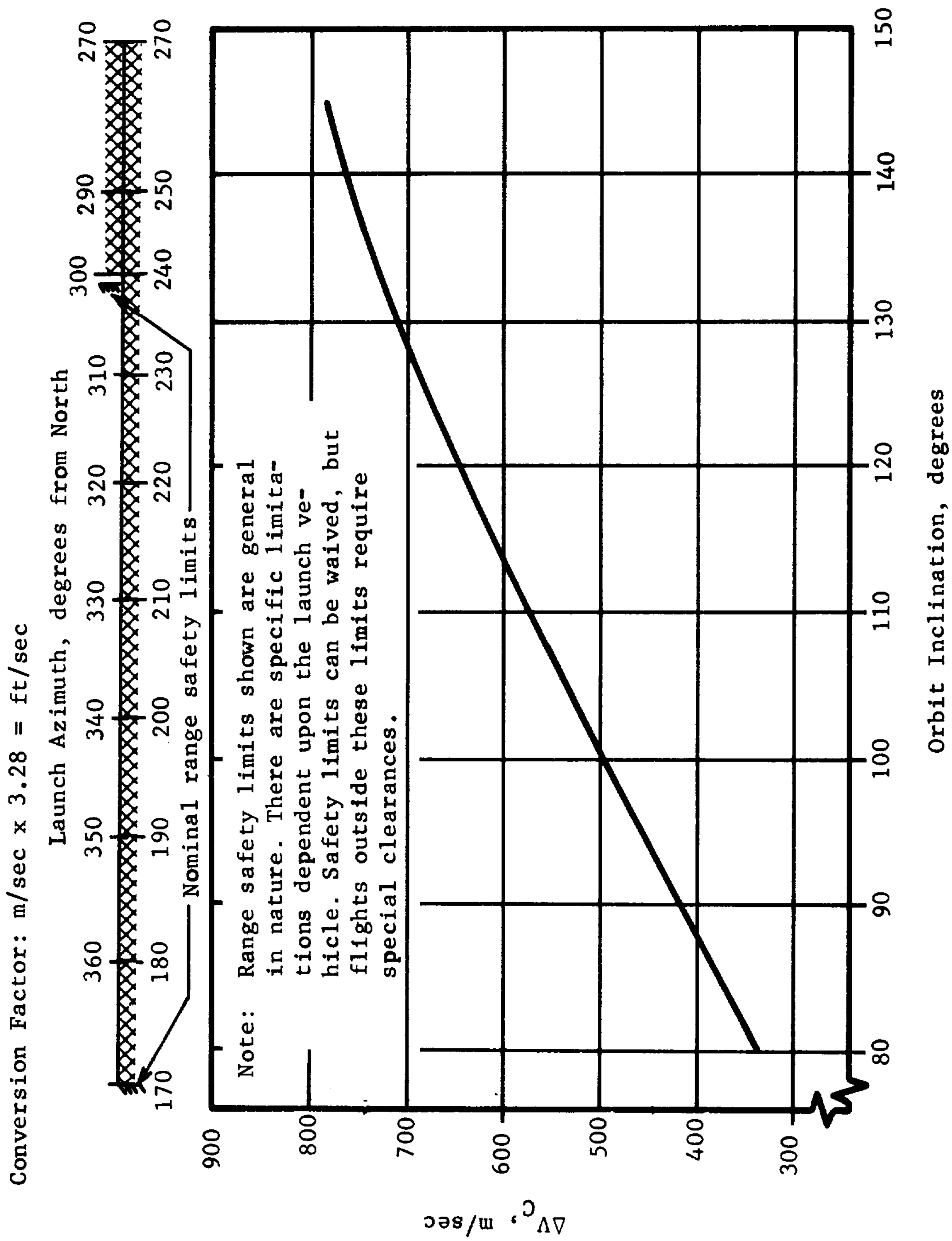


FIGURE 4-2. VELOCITY PENALTY FOR LAUNCHES FROM THE WESTERN TEST RANGE

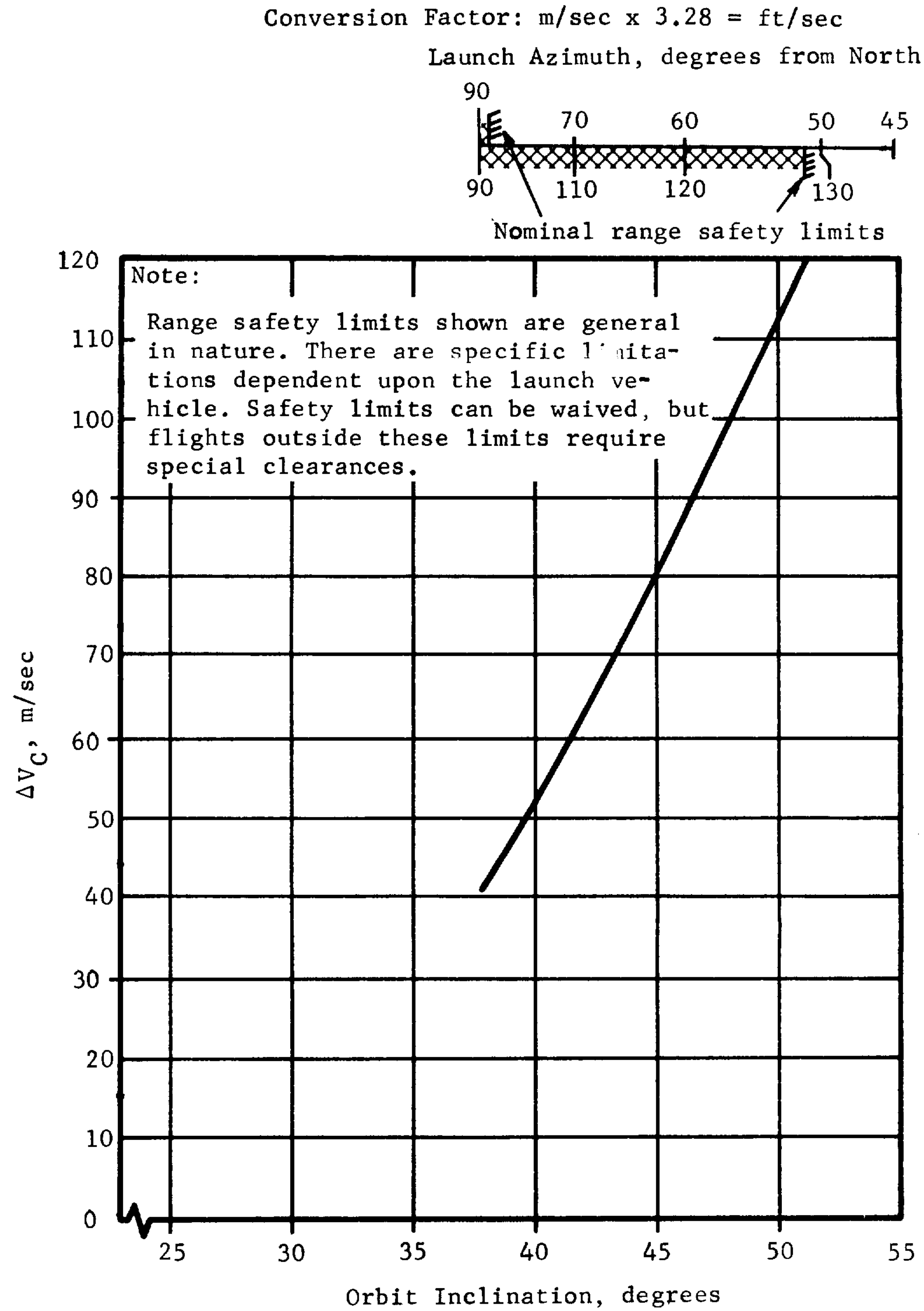


FIGURE 4-3. VELOCITY PENALTY FOR LAUNCHES FROM WALLOPS ISLAND

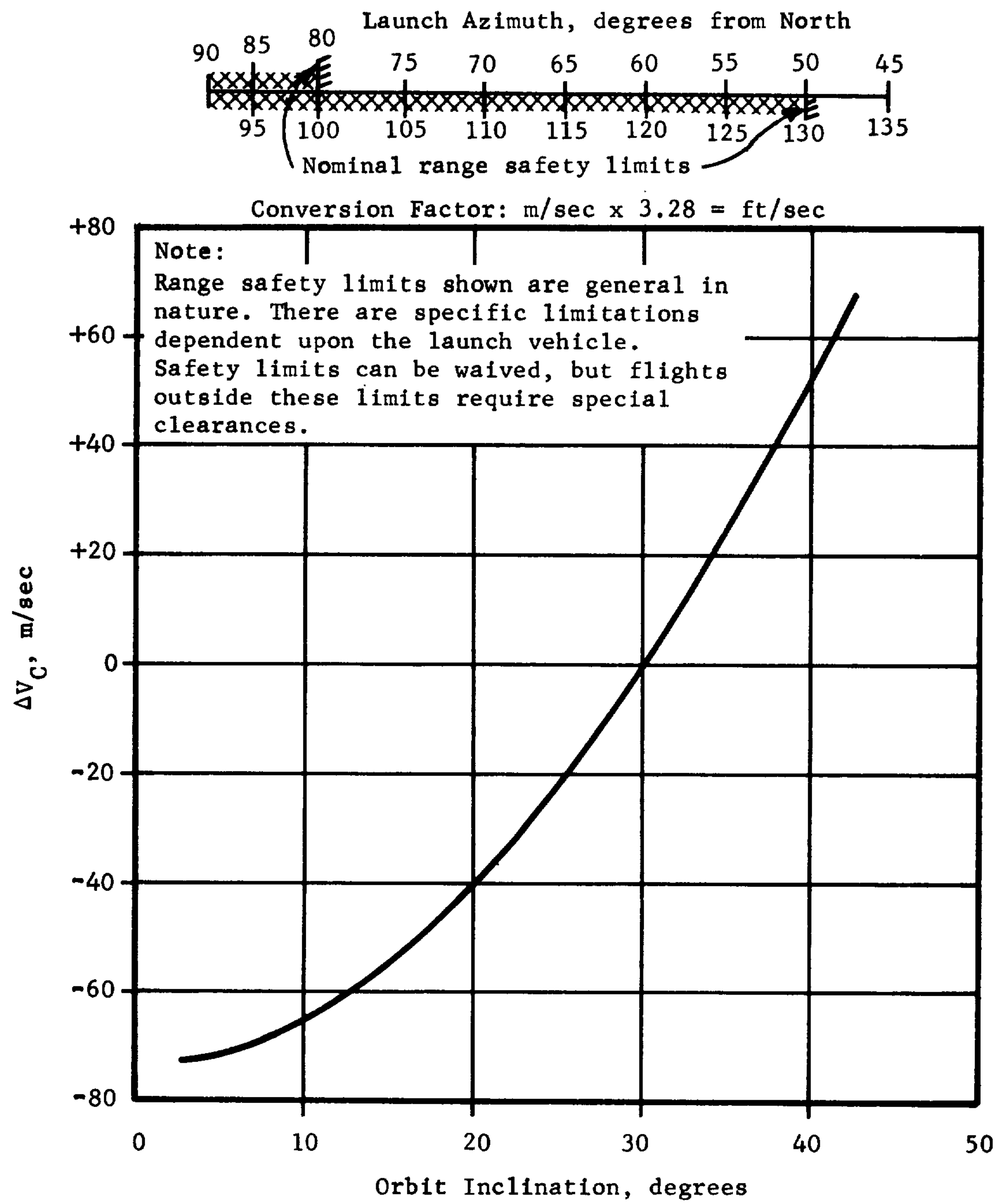


FIGURE 4-4. VELOCITY CORRECTION FOR LAUNCHES FROM SAN MARCO

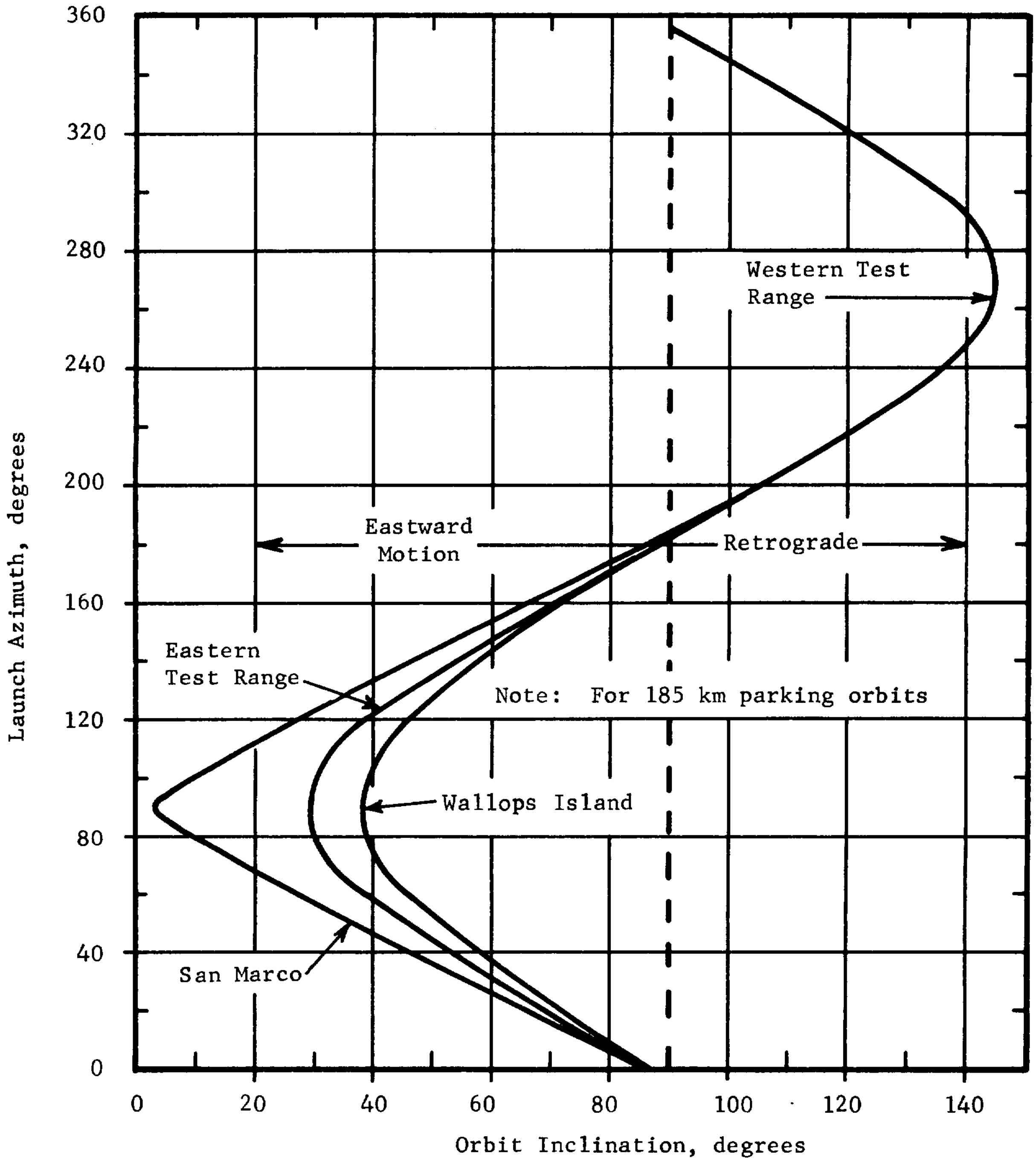


FIGURE 4-5. LAUNCH AZIMUTH REQUIRED TO ACHIEVE VARIOUS ORBIT INCLINATIONS

CHAPTER 5: GENERALIZED PERFORMANCE OF EXPENDABLE LAUNCH VEHICLES

TABLE OF CONTENTS

CHAPTER 5: GENERALIZED PERFORMANCE OF EXPENDABLE LAUNCH VEHICLES

500 INTRODUCTION

1. The figures in this chapter present performance-capability data for expendable launch vehicles using chemical propulsion. Capabilities are given in terms of payload versus characteristic velocity. When referring to launch vehicle performance, characteristic velocity is the actual total velocity deliverable for a given payload. The data presented are based on an assumed use of a 185-km circular reference parking orbit after an eastward launch from the Eastern Test Range (ETR). This definition is compatible with the definition of characteristic velocity given in Chapters 2 and 3 with regard to mission requirements provided that appropriate adjustments are made according to the procedure described in Chapter 4 for other launch sites and/or launch azimuths.

2. For launch vehicles employing chemical propulsion systems, the spacecraft payload includes all elements nominally associated with the spacecraft that must be accelerated to a required final velocity. Payload adapters may be considered either as part of the spacecraft or part of the launch vehicle. In this chapter, payload adapters are considered as part of the launch vehicle. Performance data given in this chapter were computed using allowances for representative payload adapters and shrouds as indicated in appropriate tables throughout the chapter.

3. Performance reserves have been included in all data presented in this chapter. For vehicles having only liquid-propellant upper stages, this reserve was established as 1.5 percent of the total characteristic velocity. For vehicles with solid propellant upper stages, a reserve of 0.5 percent of the velocity contribution of the solid propellant upper stage(s) and 1.5 percent of the velocity contribution of the rest of the vehicle was assumed. Scout vehicles were treated separately. For these, a reserve of 0.5 percent of the total characteristic velocity was assumed.

4. Performance data for only a few of the possible expendable chemical propulsion launch vehicles that could be postulated are shown. Those vehicles that are included were selected as reasonable

alternatives on the basis of the requirements for improved performance as indicated by analysis of recently proposed missions, analysis of expected costs, and anticipated difficulty of development. The selections should be regarded as constituting the best current estimate of likely candidates from which a family of expendable launch vehicle can be chosen for OSS and OA applications over the next 20 years.

5. Data on these expendable launch vehicles are provided by the Director of Launch Vehicle and Propulsion Programs for use by OSS, OA, DOD, and other mission planners. In all cases, the data represent the best estimates available for mission applications of interest in advance planning.

501 PROCEDURE

1. The performance data in this chapter may be used in the following manner. For solar-system missions, the curves in Chapter 2 can be used to convert the desired destination and flight time into a required characteristic velocity for the mission. For Earth-orbital missions, the curves in Chapter 3 can be used in the same manner to convert the desired apogee and perigee into a required characteristic velocity. For all missions that are to be launched due east from the ETR, these characteristic velocities can be used directly with the curves in this chapter to obtain the payload deliverable by various launch vehicles. If the mission is to be launched at some other launch azimuth or from one of the other launch sites, the characteristic velocity obtained from Chapters 2 and 3 must be modified by a correction obtained from Chapter 4. This modified characteristic velocity can then be used with the curves in this chapter to obtain payload capabilities for various expendable launch vehicles. It should be remembered that the weight of any velocity packages, kick motors, or other systems necessary to accomplish the mission must be added to the basic spacecraft weight to determine the total payload which must be added to the basic spacecraft weight to determine the total payload which must be delivered by the launch vehicle.

502 CAUTIONARY NOTE

1. The mission planner is warned against the indiscriminate use of the generalized performance curves in this subsection for estimating launch-vehicle capabilities for Earth-orbital missions. Specifically, the Earth-orbital performance of vehicles which use

solid propellant or other nonrestartable final stages (such as Burner II or Core II on Titan IIID) cannot be obtained by the above procedure. For Scout, Delta (various configurations), SLV3D/ Centaur (single burn), TIIIC (single burn Transtage), and TIIID configurations, the curves in Chapter 6 must be used in estimating capabilities for Earth-orbital missions.

503 CURRENT LAUNCH VEHICLES

1. Table 5-1 provides a listing of expendable launch vehicles that are presently available or that will be available in the very near future. Group 1 consists of small and intermediate launch vehicles, while the larger vehicles are included in Group 2.

2. The Scout vehicles that appear are uprated versions that use an Algol III solid propellant first stage. Scout D is a four-stage vehicle, and Scout E has a fifth stage for higher energy missions.

3. The Delta vehicles are identified by a four-digit numerical designation. The first digit identifies the Thor booster configuration [currently a two (2) identifies the "straight eight" Thor]. The second digit represents the number of Castor II solid rocket motors that are strapped to the Thor booster for thrust augmentation. The third digit identifies the second stage being used [a one (1) for the Delta stage configured for the "straight eight" vehicles]. The fourth digit is zero (0) for all two-stage vehicles. For three-stage vehicles, a three (3) indicates that the vehicle has a TE364-3 (1440) third stage, and a four (4) indicates that the third stage is a TE364-4 (2300).

4. The SLV3D/Centaur vehicle is the Atlas booster with the Centaur D-1A upper stage.

5. Titan IIIB is the basic two-stage Titan core vehicle which includes the Core I and Core II stages. The abbreviation (Str. CI) indicates a stretched version of Core I. The second stage, Core II, does not have restart capability. Titan IIID employs the basic core with two five-segment 120-inch-diameter solid rocket motor strapons as a "zero" stage. Titan IIIC uses the Titan Transtage as an upper stage on the Titan IIID stages. Titan IIIE is a Titan IIID modified for use with the Centaur stage.

6. Saturn IB consists of the SIB booster with an SIVB upper stage; it is sometimes designated as the uprated Saturn I.

7. Scout D and E, SLV3D/Centaur/TE364(2300), Delta Models 2313, 2613, 2913, 2314, 2614, 2914, and Titan IIIE/Centaur/TE364-4

(2300) employ spin-stabilized final stages. Hence, payloads for these vehicles must be capable of withstanding the spin and must either operate while spin-stabilized or have appropriate despin devices.

TABLE 5-1. LAUNCH VEHICLES AVAILABLE IN 1973-1978

Launch Vehicle	Payload Adapter Mass[(a)], kg	Shroud Mass[(a)], kg	Shroud Configuration
Group 1 (See Figure 5-1)			
Scout D	5.5	120	Scout
Scout E	1.5	127	Scout, extended
Delta Model 2310	20	550	Straight 8
Delta Model 2610	30	550	Straight 8
Delta Model 2910	40	550	Straight 8
Delta Models 2314, 2614, and 2914	23	550	Straight 8
Group 2 (See Figure 5-2)			
Titan IIIB(Str. CI)/Agena	45	1050	Titan(UPLF)
SLV3D/Centaur	53	925	D-1A
SLV3D/Centaur/TE364-4(2300)	9	925	D-1A
Titan IIIC	36	900	Titan(UPLF)
Titan IIID	36	2720	Centaur Standard
Titan IIIE/Centaur	53	2720	Centaur Standard
Titan IIIE/Centaur/TE364-4(2300)	9	2720	Centaur Standard
Saturn IB	700	2500	Conceptual
Saturn V	700	2500	Conceptual

(a) Performance data in this chpater are given with adjustments for payload adapters and shrouds as indicated. Conversion factor: kg x 2.20 = lb.

504 POSSIBLE NEAR-TERM LAUNCH VEHICLES

1. Table 5-2 lists improved launch vehicles that may possibly be available in the near future. Group 1 includes improved versions of Scout, and various applications of the Burner II upper stages. BII(1440) is a Burner II stage built around a TE364-3(1440) solid propellant rocket motor, and BII(2300) is a Burner II stage built around a TE364-4(2300) solid propellant rocket motor. The performance curves for this group of proposed launch vehicles are shown on Figure 5-3. Group 2 includes the SLV3D/Delta(TSE),

SLV3D/Centaur/BII(2300), and various Titan configurations. Performance curves for Group 2 launch vehicles are shown on Figure 5-4.

TABLE 5-2. 1973-1978 IMPROVED LAUNCH VEHICLE POSSIBILITIES

Launch Vehicle	Payload Adapter Mass[(a)], kg	Shroud Mass[(a)], kg	Shroud Configuration
	Group 1 (See Figure 5-3)		
ASLV	58	414	Conceptual
ASLV (with Kick motor)	58	414	Conceptual
TAT(3C)/BII(1440)	9	125	Burner II
TAT(9C)/BII(1440)	9	125	Burner II
TAT(9C)/BII(2300)	9	300	Burner II
SLV3A/BII(2300)	9	380	Atlas/BII
TAT(3C)/Agena	45	230	Agena Long Shell
TAT(9C)/Delta/BII(2300)	23	550	Straight 8
	Group 2 (See Figure 5-4)		
Titan IIIB(Str. CI)/Delta(TSE)	40	250	Delta
SLV3D/Centaur/BII(2300)	20	925	Surveyor
SLV3D/Delta(TSE)	40	250	Delta
Titan IIIB/Centaur	53	2720	Viking
Titan IIIB(Str. CI + 2A3)/Centaur	53	2720	Viking
Titan III7	36	2720	Viking
Titan IIIC7	36	896	Titan(UPLF)
Titan III7/Centaur	55	2720	Viking

(a) Performance data in this chapter are given with adjustments for payload adapters and shrouds as indicated. Conversion factor: kg x 2.20 = lb.

2. NASA is currently considering an improved version of the Scout Launch Vehicle with increased performance. This vehicle is referred to as the Advanced Small Launch Vehicle (ASLV). Another proposed method of launching small payloads is to mate Burner II upper stages with the TAT(nC) boosters: TAT is the Thrust Augmented Thor/Delta booster, and n is the number of Castor II solid rocket motor strapons used for the thrust augmentation. The proposed Titan IIIB(Str. CI + 2A3) consists of a stretched Core I with two Algol III solid rocket motor strapons (for thrust

augmentation) and a standard Core II (Core II does not have restart capability). The Titan IIIC7 and Titan III7 both use the stretched Core I with 7-segment, 120-inch-diameter solid rocket motor strapons used as a "zero" stage. The Titan IIIC7 would also have a stretched version of the Transtage, sized to provide optimal tankage for synchronous equatorial missions.

505 CONCEPTUAL SATURN-CLASS LAUNCH VEHICLES

1. Table 5-3 lists a few conceptual Saturn-class launch vehicles. Performance data for these vehicles are shown in Figure 5-5. Solid propellant boosters could be constructed by clustering 156-inch-diameter solid rocket motors within an appropriate structure. The 4 x 1563 booster is a cluster of four, three-segment 156-inch-diameter motors. The SIVB(J2S) would be powered by the J2S engine, which is a proposed modification to the standard J2 engine.

TABLE 5-3. CONCEPTUAL SATURN-CLASS LAUNCH VEHICLES

(See Figure 5-5)

Launch Vehicle	Payload Adapter Mass[(a)], kg	Shroud Mass[(a)], kg	Shroud Configuration
4 x 1563/SIVB(J2S)	700	3700	Conceptual
Saturn IB/Centaur	136	2500	Conceptual
4 x 1563/SIVB(J2S)/Centaur	136	3700	Conceptual
Saturn V/Centaur	136	2500	Conceptual

(a) Performance data in this chapter are given with adjustment for payload adapters and shrouds as indicated. Conversion factor: kg x 2.20 = lb.

506 TITAN VEHICLE OPTIONS FOR INTERPLANETARY MISSIONS

1. Table 5-4 lists a variety of Titan launch-vehicle options for interplanetary missions. Figure 5-6 shows the performance of these Titan-based vehicles with the Burner II(2300) velocity package. The Centaur GT is a proposed modified Centaur that would have a propellant capacity of approximately 20,400 kg (45,000 lb).

TABLE 5.4. TITAN VEHICLE OPTIONS FOR INTERPLANETARY MISSIONS

(See Figure 5-6)

Launch Vehicle	Payload Adapter Mass[(a)], kg	Shroud Mass[(a)], kg	Shroud Configuration
Titan IIIC/BII(2300)	9	900	Titan (UPLF)
Titan IIIE/Centaur/BII(2300)	9	2720	Viking
Titan IIIE/Centaur GT/BII(2300)	9	2720	Viking
Titan IIIC7/BII(2300)	9	900	Titan (UPLF)
Titan III7/Centaur/BII(2300)	9	2720	Viking
Titan III7/Centaur GT/BII(2300)	9	2720	Viking

(a) Performance data in this chapter are given with adjustments for payload adapters and shrouds as indicated. Conversion factor: kg x 2.20 = lb.

507 LAUNCH VEHICLES WITH POSSIBLE ADVANCED UPPER STAGES

1. Table 5-5 lists various launch vehicles with a possible Versatile Upper Stage (VUS). Performance data are shown on Figure 5-7. The Versatile Upper Stage is a concept studied recently by NASA. The performance shown here is based on the results of an advance study of the stage assuming liquid hydrogen and oxygen propellants.

TABLE 5-5. SELECTED LAUNCH VEHICLES WITH A POSSIBLE H_2/O_2 VERSATILE UPPER STAGE (VUS)

(See Figure 5-7)

Launch Vehicle	Payload Adapter Mass[(a)], kg	Shroud Mass[(a)], kg	Configuration
TAT(9C)/VUS	30	550	Straight 8
TAT(9C)/VUS/TE364-4(2300)	23	550	Straight 8
SLV3D/VUS	30	450	New
Titan IIIB(Str. CI)/VUS	30	450	New
Titan IIIE/Centaur/VUS	30	2720	Viking

(a) Performance data in this chapter are given with adjustments for payload adapters and shrouds as indicated. Conversion factor: kg x 2.20 = lb.

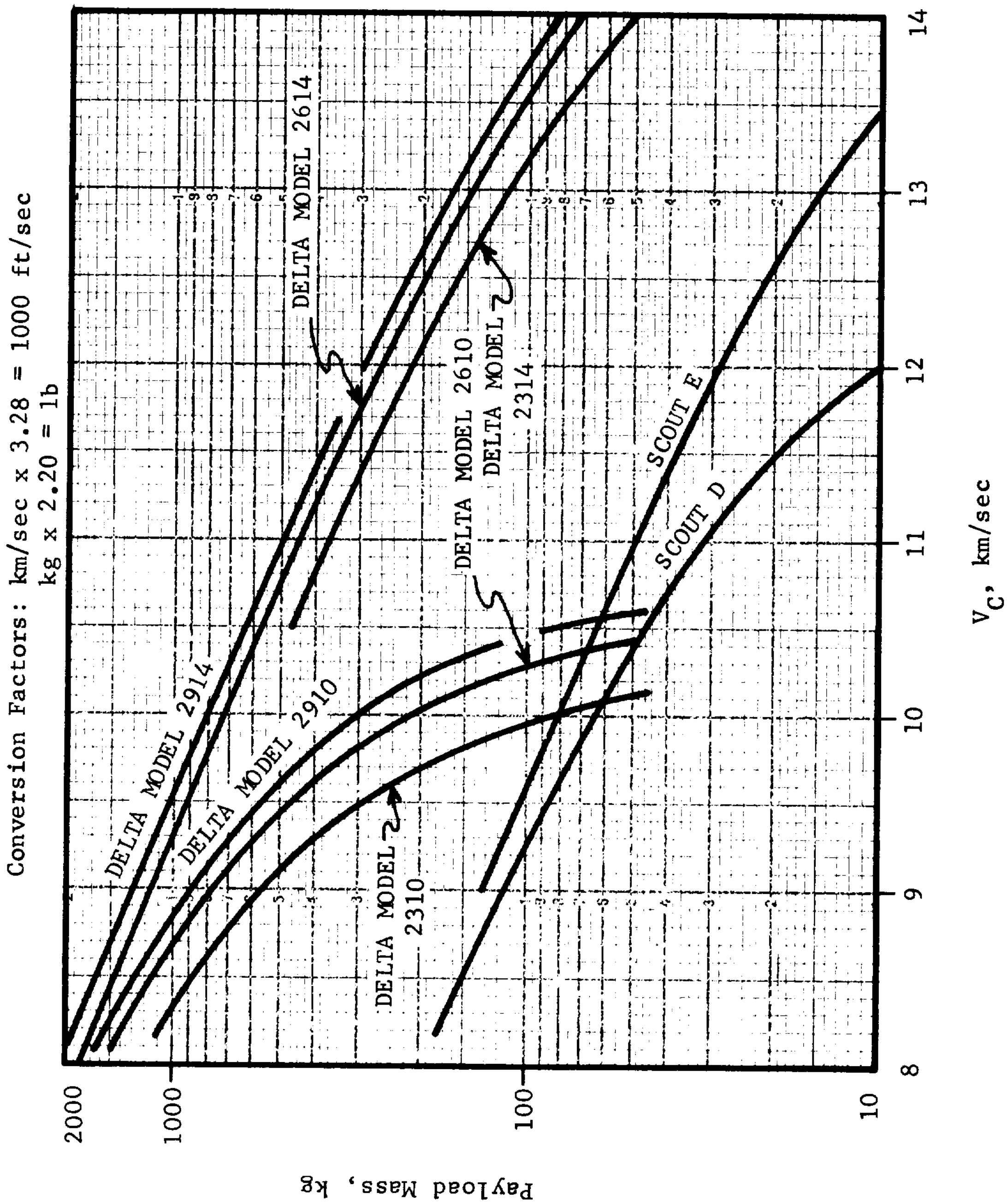


FIGURE 5-1. LAUNCH VEHICLE PERFORMANCE AVAILABLE IN 1973-1978 (GROUP 1)

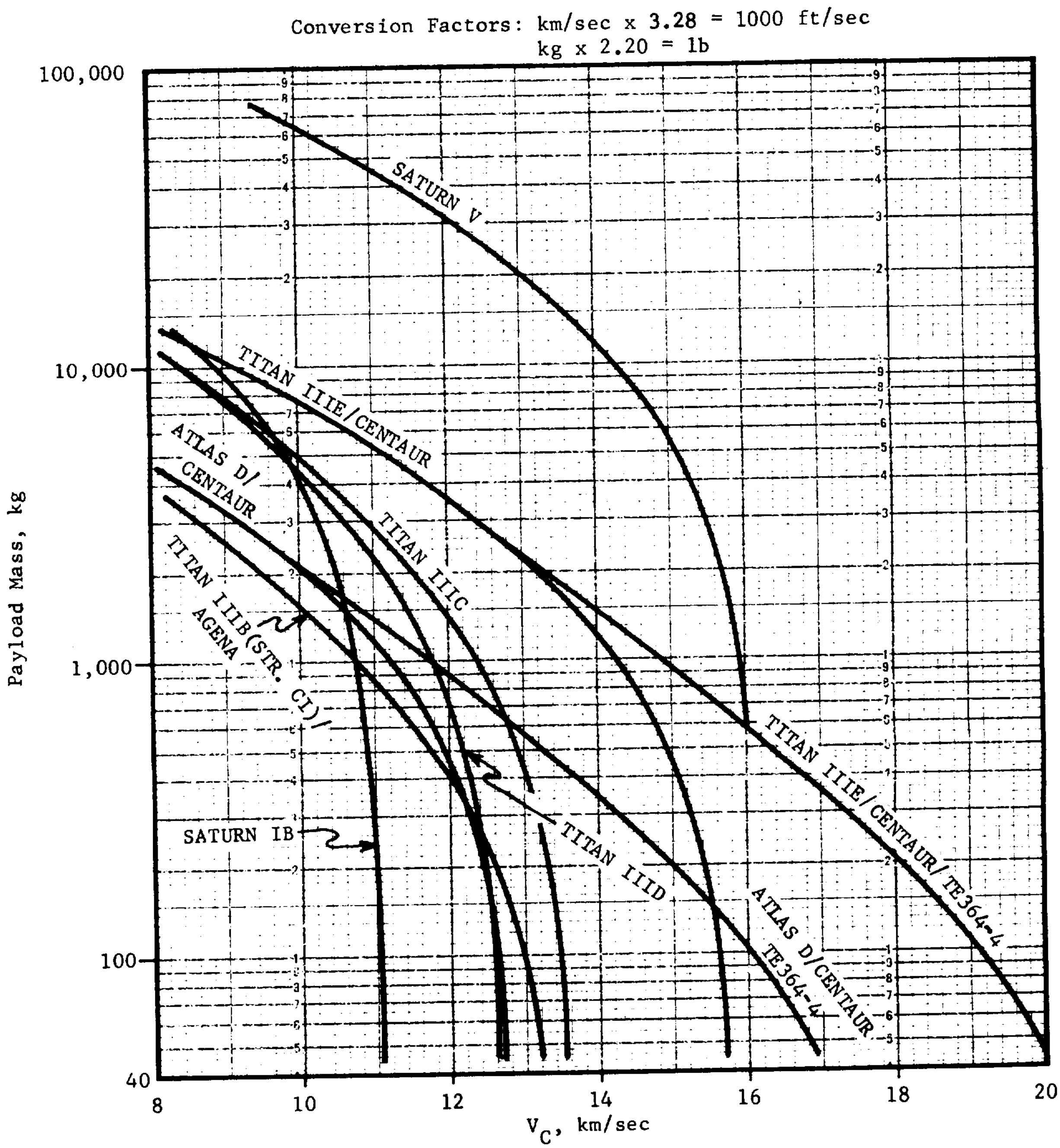


FIGURE 5-2. LAUNCH VEHICLE PERFORMANCE AVAILABLE IN 1973-1978 (GROUP 2)

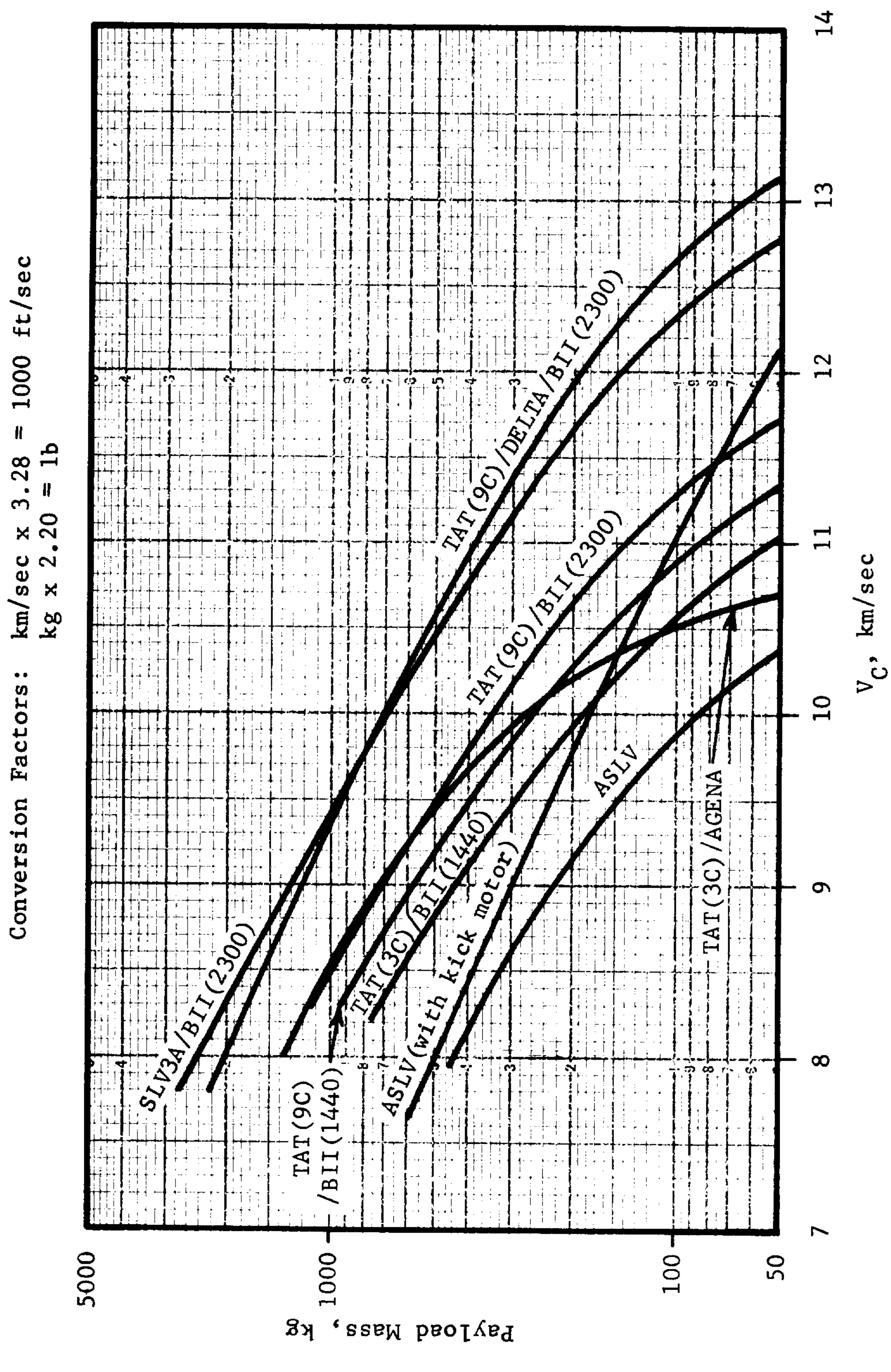


FIGURE 5-3. POSSIBLE NEAR-TERM LAUNCH VEHICLES (GROUP 1)

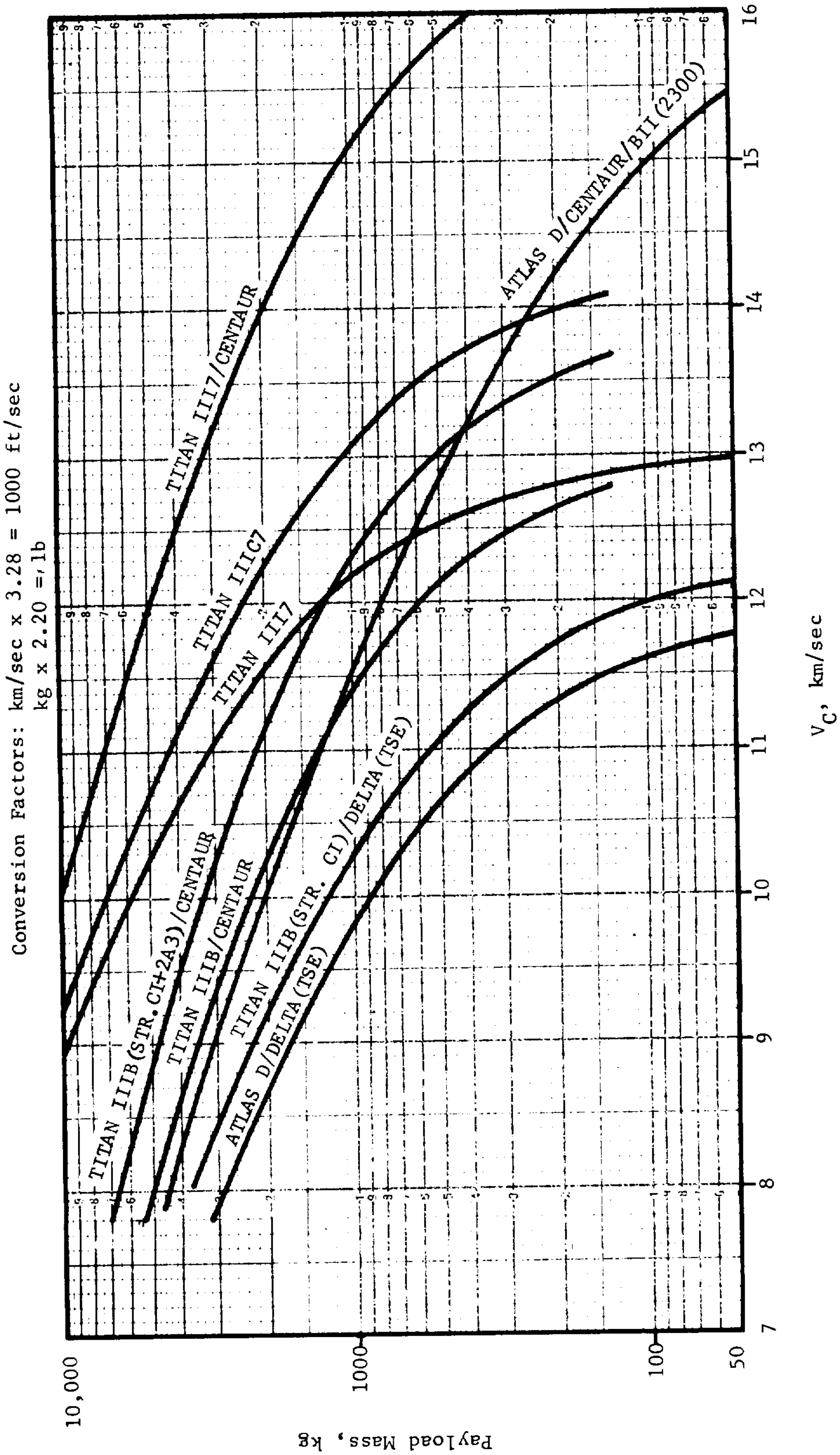


FIGURE 5-4. POSSIBLE NEAR-TERM LAUNCH VEHICLES (GROUP 2)

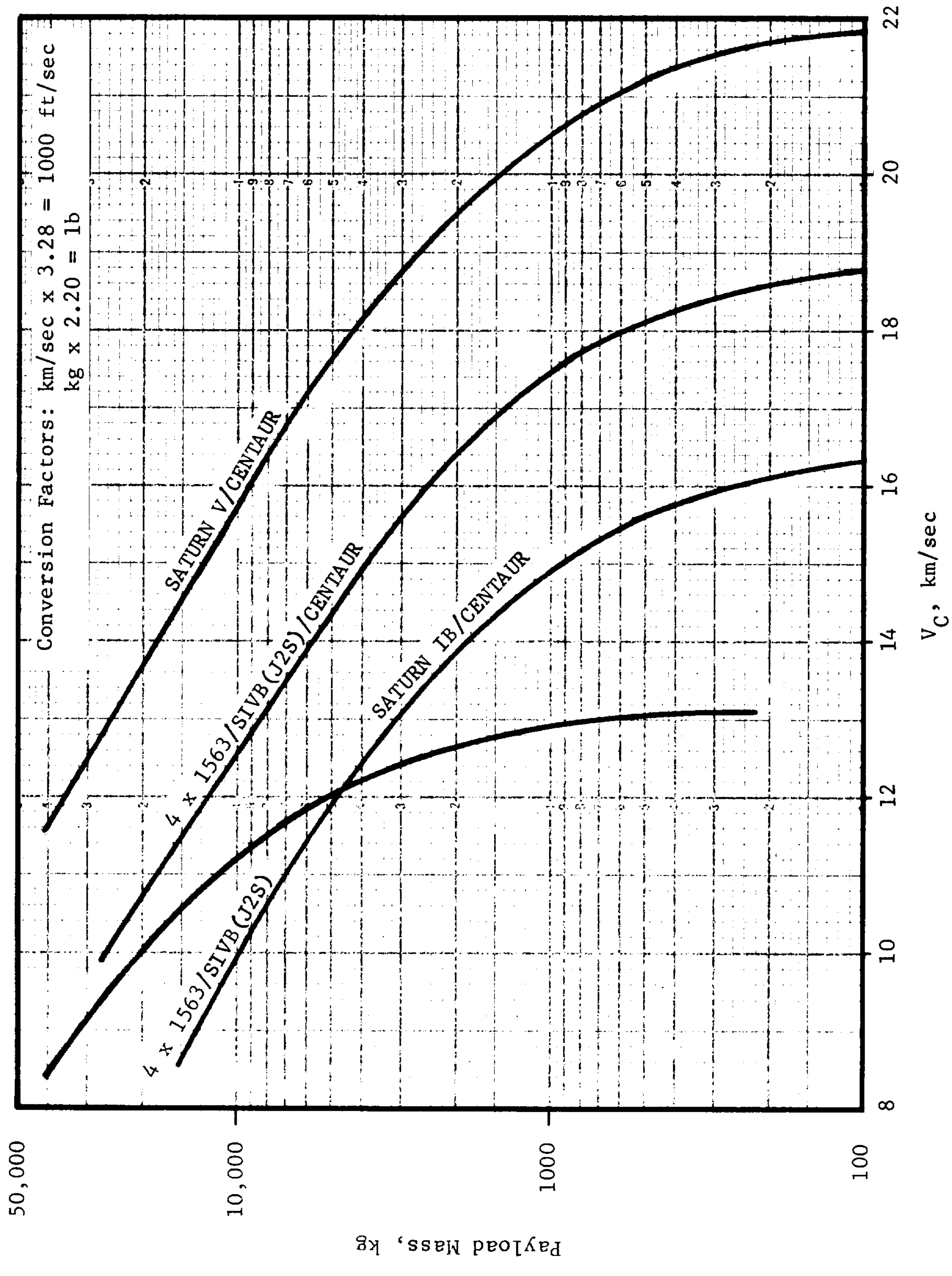


FIGURE 5-5. CONCEPTUAL SATURN CLASS LAUNCH VEHICLES

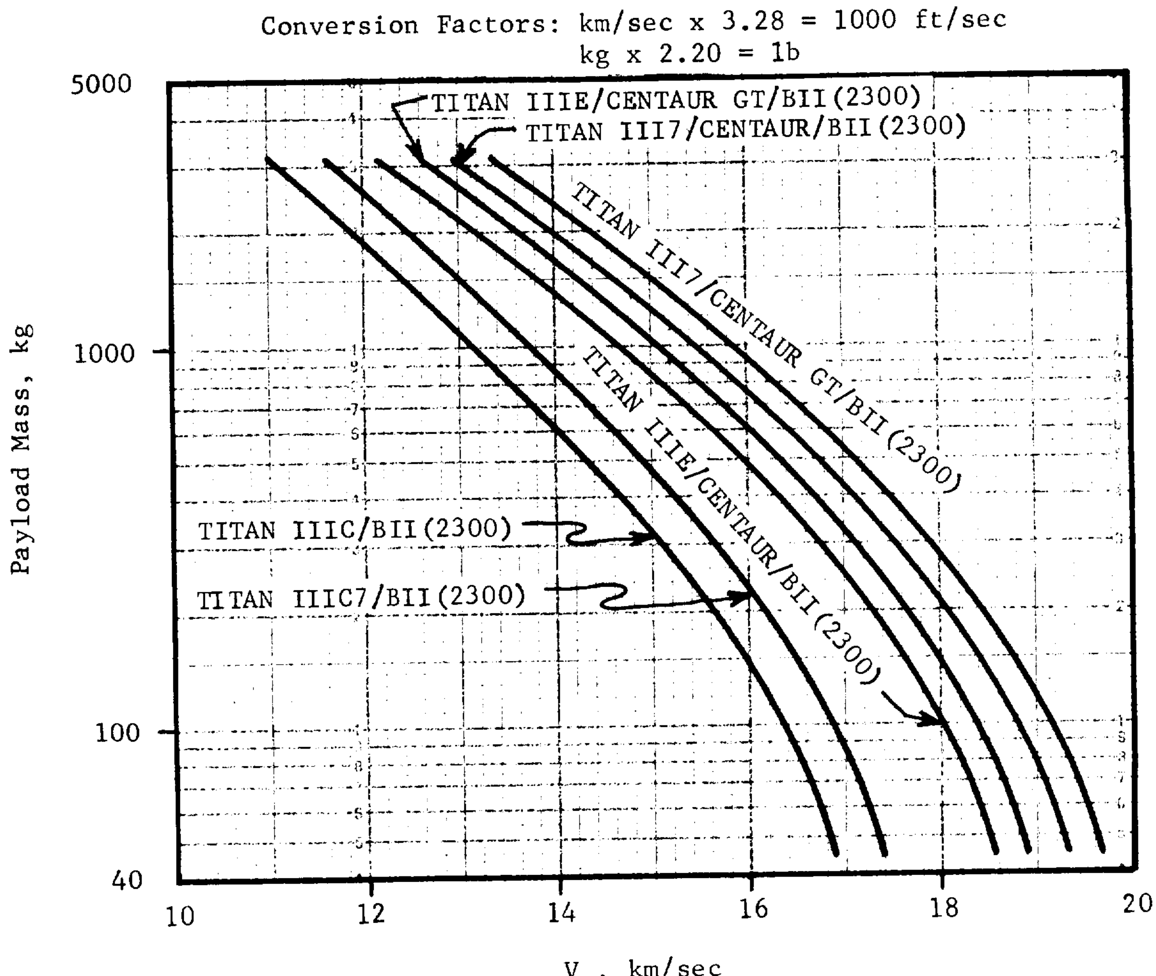


FIGURE 5-6. PERFORMANCE OF VARIOUS TITAN VEHICLE OPTIONS FOR INTERPLANETARY MISSIONS

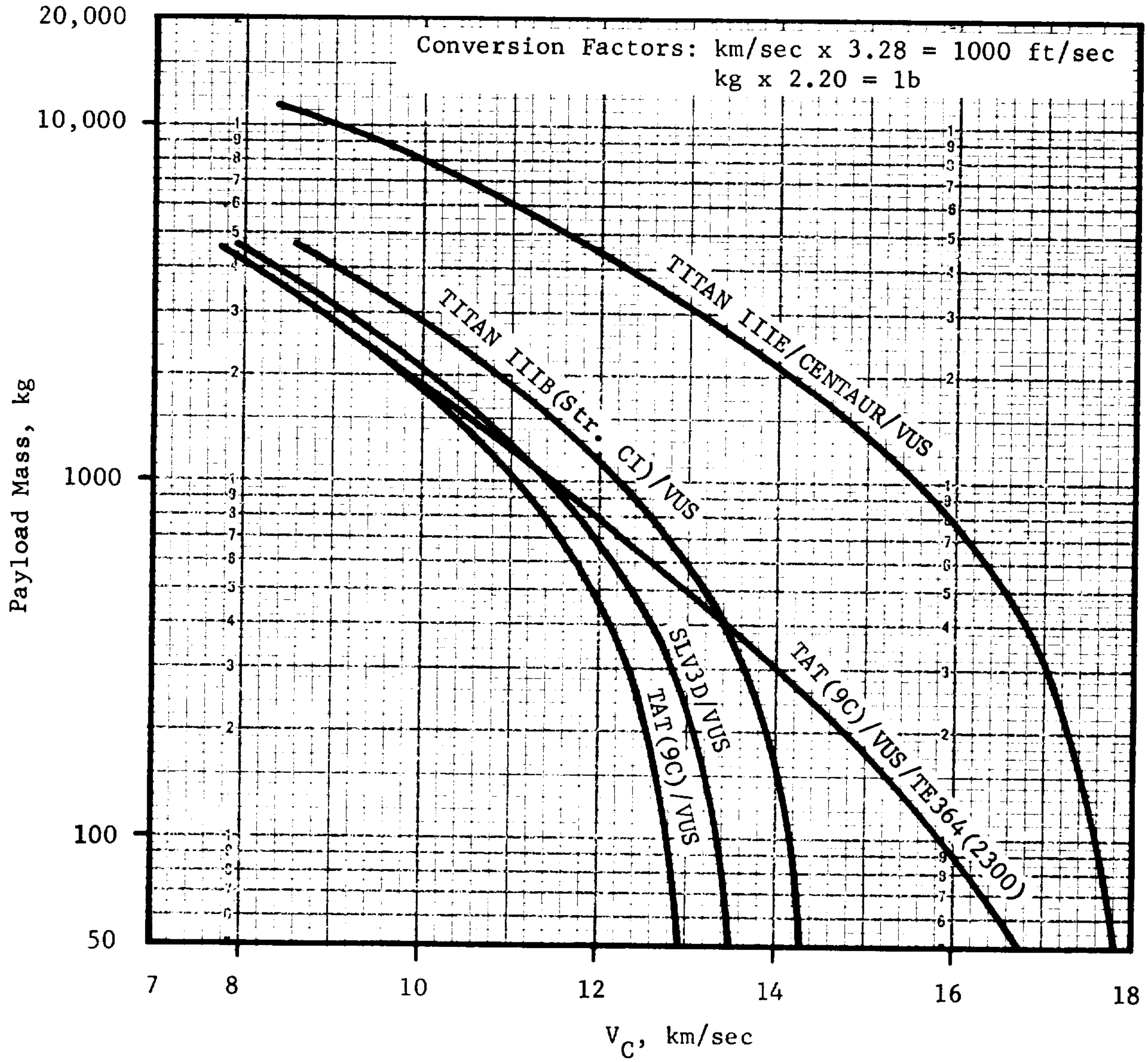


FIGURE 5-7. PERFORMANCE OF SELECTED LAUNCH VEHICLES WITH A POSSIBLE H_2/O_2 VERSATILE UPPER STAGE (VUS)

CHAPTER 6: EARTH-ORBITAL PERFORMANCE OF EXPENDABLE LAUNCH VEHICLES

TABLE OF CONTENTS

CHAPTER 6: EARTH-ORBITAL PERFORMANCE OF EXPENDABLE LAUNCH VEHICLES

600 GENERAL

1. Because of constraints imposed on stage restart, coast time, guidance, and other limitations, it is necessary to present separate data for estimating the Earth orbital performance of Scout, TAT/Delta (various configurations), SLV3D/Centaur (single burn), Titan IIIC (single burn), and Titan IIID launch vehicle configurations.

2. In all cases, the curves in this chapter represent the best data available at the date of publication. Unlike the data in Chapter 5, no allowance has been made for payload adapters. The payload values read from the graphs in this chapter must include the weight of the spacecraft adapter. Range safety considerations (such as the impact point of lower stages) may cause the Earth-orbital performance of some vehicles to be less than that given in this chapter. For information on this subject as well as for estimates on orbital payload capabilities for other vehicles with nonrestartable last stages, or other launch azimuths or perigee values, contact one of the persons listed in the Preface.

601 SCOUT LAUNCH VEHICLES

1. Payload capabilities for the four-stage Scout D are shown in Figures 6-1 through 6-5. Figure 6-1 is for launches due east from Wallops Island, Figure 6-2 is for launches due east from San Marco, and Figure 6-3 shows polar orbit capabilities from the WTR. Figures 6-4 and 6-5 show orbital payload capability for the five-stage Scout E for eastward launches from Wallops Island and polar orbits from the WTR, respectively. These launch vehicles are uprated versions of Scout which employ an Algol III first stage. All of the Scout performance data shown in these figures are based on the standard 86.36 cm (34 inch) diameter Scout shroud. An extended standard shroud and a 106.86 cm (42 inch) diameter shroud are also available (see Chapter 12). When the larger shrouds are used, the payload that the Scout vehicles can place in orbit is reduced by approximately 6 percent. (See Figures 6-1, 6-2, and 6-3.)

602 TAT/DELTA LAUNCH VEHICLES

1. TAT/Delta/TE364-3 (1440) and TAT/Delta/TE364-4 (2300) data are shown for elliptic orbits with a 185 km perigee in Figures 6-6, 6-7, and 6-8. Figures 6-6 and 6-7 show the performance for due-east launches from the ETR, and Figure 6-8 shows the performance for polar launches from the WTR. These vehicles are limited in the range of payloads they can launch because the Delta stage must be able to achieve a parking orbit. This restriction can be waived; however, special arrangements must be made to comply with range safety and argument of perigee requirements.

2. In performing synchronous transfer missions with three-stage TAT/Delta vehicles, the TE364 can do part of the plane change required for a synchronous equatorial orbit from ETR. Figure 6-9 shows payload as a function of transfer orbit inclination for the family of three-stage Delta vehicles.

3. The TAT/Delta vehicles are identified by a four-digit numerical designation. The first digit identifies the Thor booster configuration. For the vehicles shown here, the two (2) indicates a stretched Thor, uprated by replacing the current MB-3 engine with a more powerful H1 engine (flown on the Saturn IB first stage). This stage is scheduled to become operational in 1973. The second digit represents the number of Castor II solid-rocket motors that are strapped to the Thor booster for thrust augmentation. The third digit is always a one (1) (for the vehicles shown) and identifies the second stage as the Delta modified to integrate with the H1 Thor and an 8-foot-diameter payload fairing. This fairing enshrouds the payload and all stages above the Thor (which is also 8 feet in diameter). Hence, these vehicles have been named the "straight eight" Deltas. The fourth digit is zero (0) for all two-stage vehicles. For three-stage vehicles, a three (3) indicates that the vehicle has a TE364-3 (1440) third stage, and a four (4) indicates that the third stage is a TE364-4 (2300).

603 ATLAS AND TITAN LAUNCH VEHICLES

1. Figure 6-10 shows the orbital capability of the SLV3D/Centaur for due-east launches from the ETR. Figure 6-11 shows the orbital capability for the Titan IIIC with Operational Transtage for direct ascent launches from the ETR. Figure 6-12 shows the orbital capability for the Titan IIID for launches from the ETR; Figure 6-13 shows the polar orbit capability for the Titan IIID from the WTR.

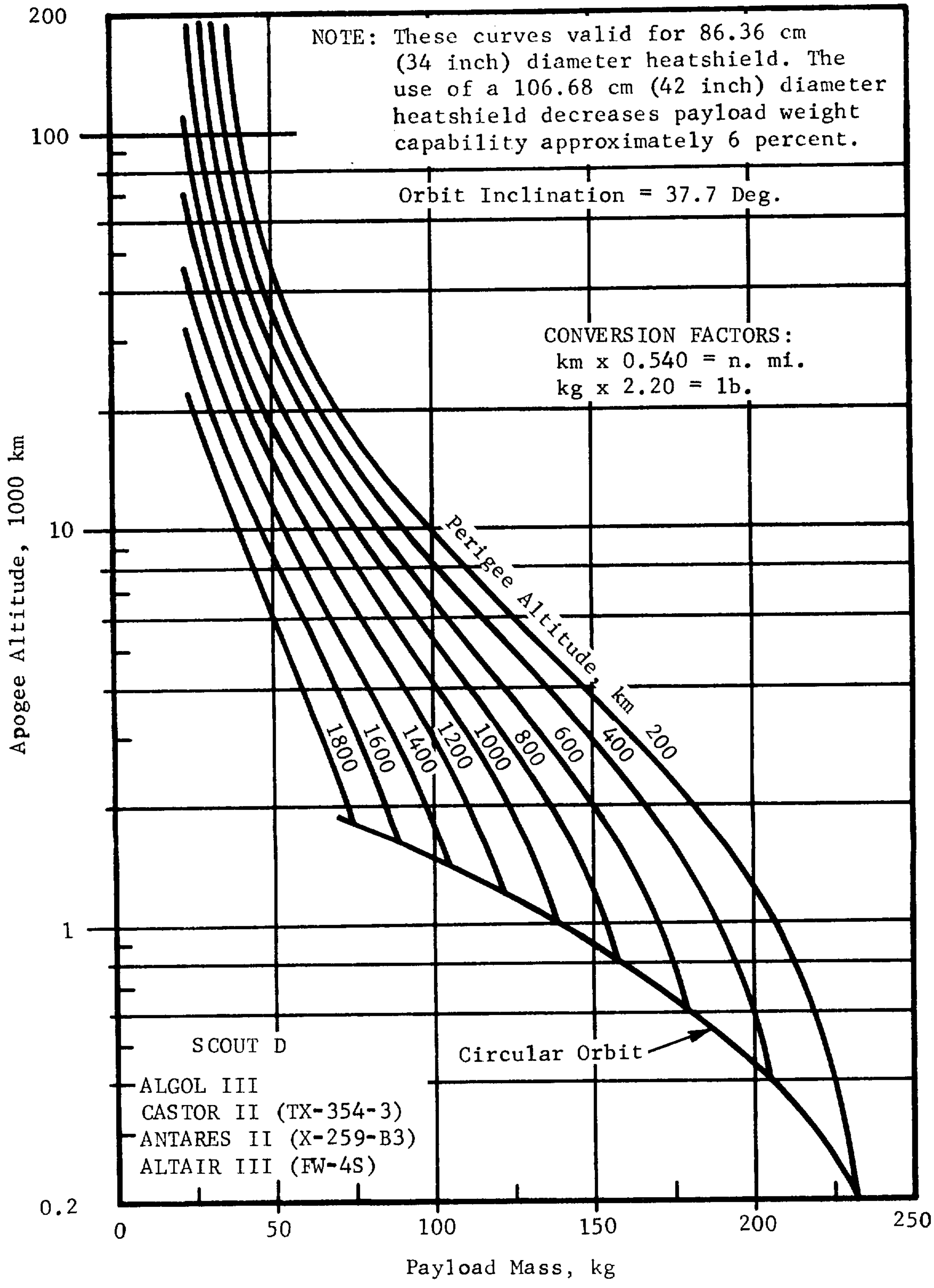


FIGURE 6-1. SCOUT ORBITAL CAPABILITY-DUE EAST, WALLOPS

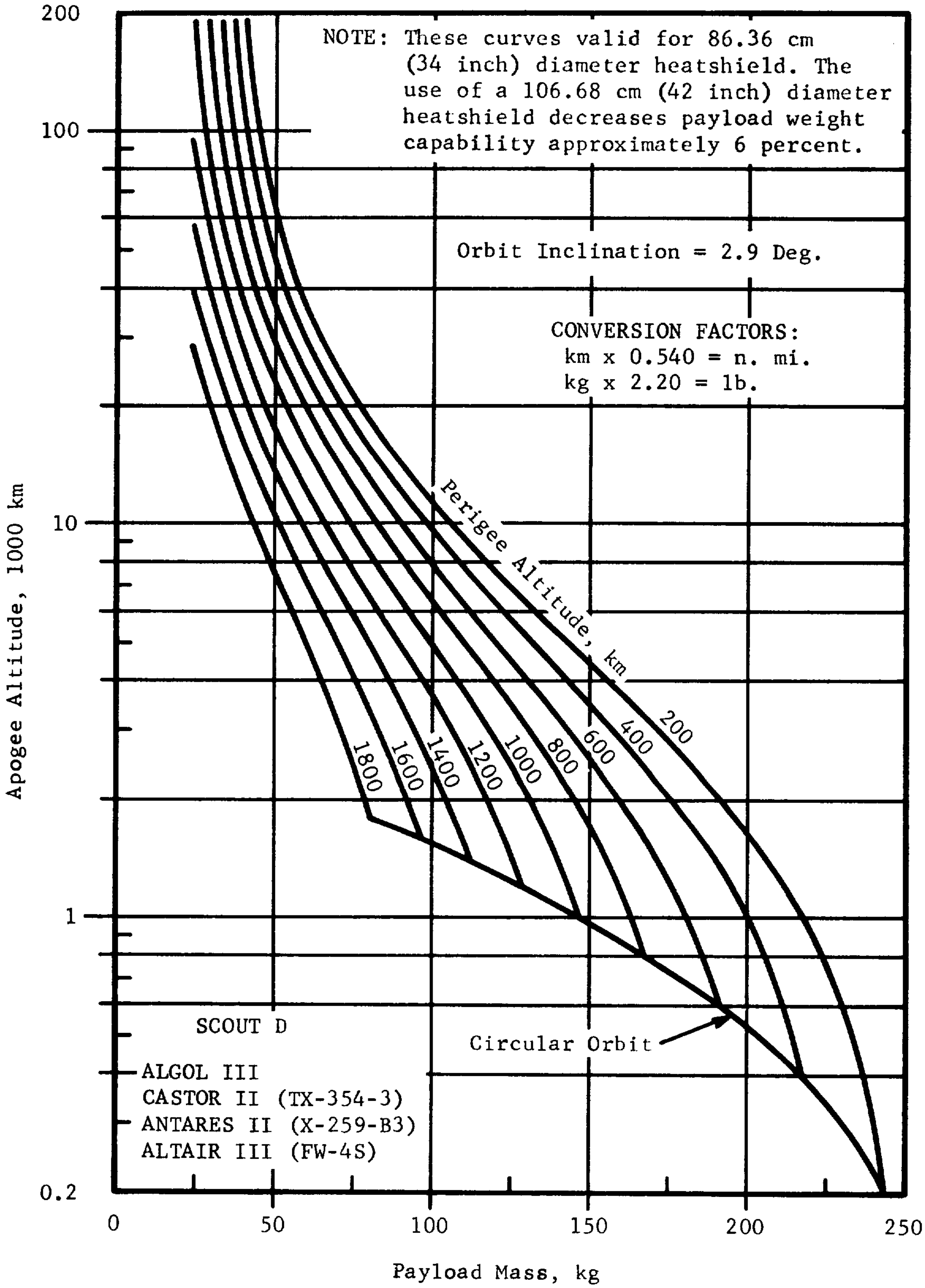


FIGURE 6-2. SCOUT ORBITAL CAPABILITY-DUE EAST, SAN MARCO

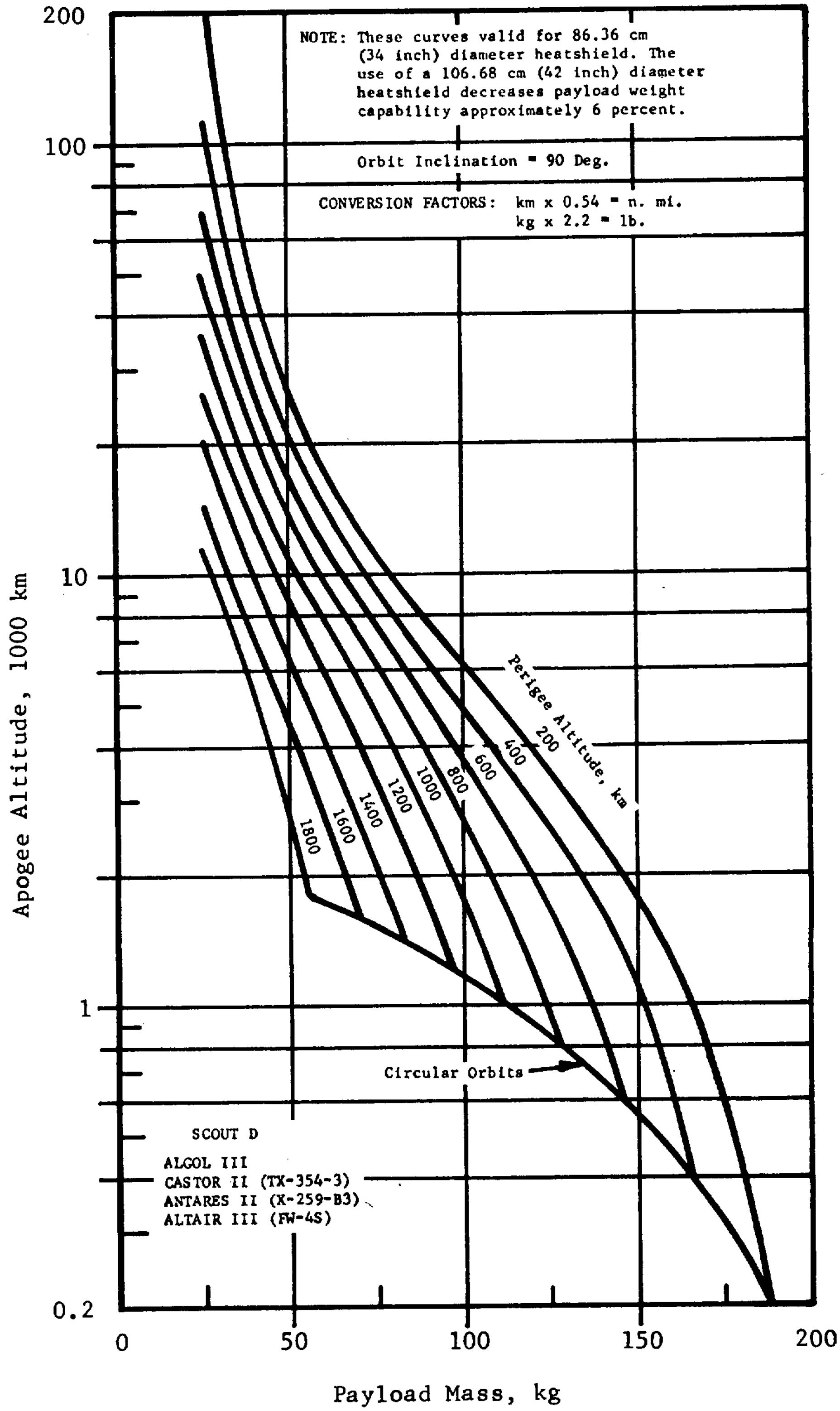


FIGURE 6-3. SCOUT POLAR ORBITAL CAPABILITY, WTR

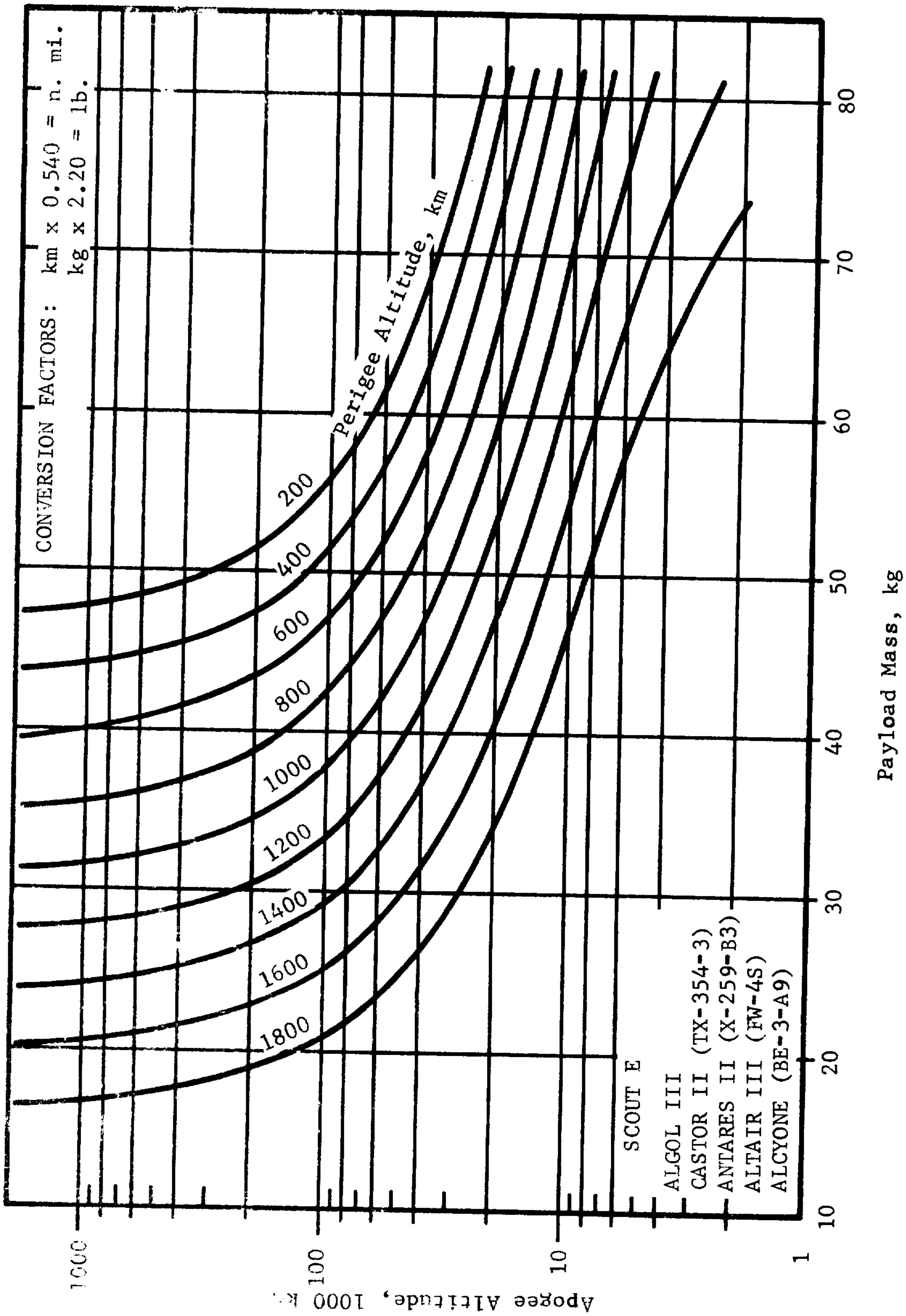


FIGURE 6-4. FIVE-STAGE SCOUT ORBITAL CAPABILITY-DUE EAST, WALLOPS

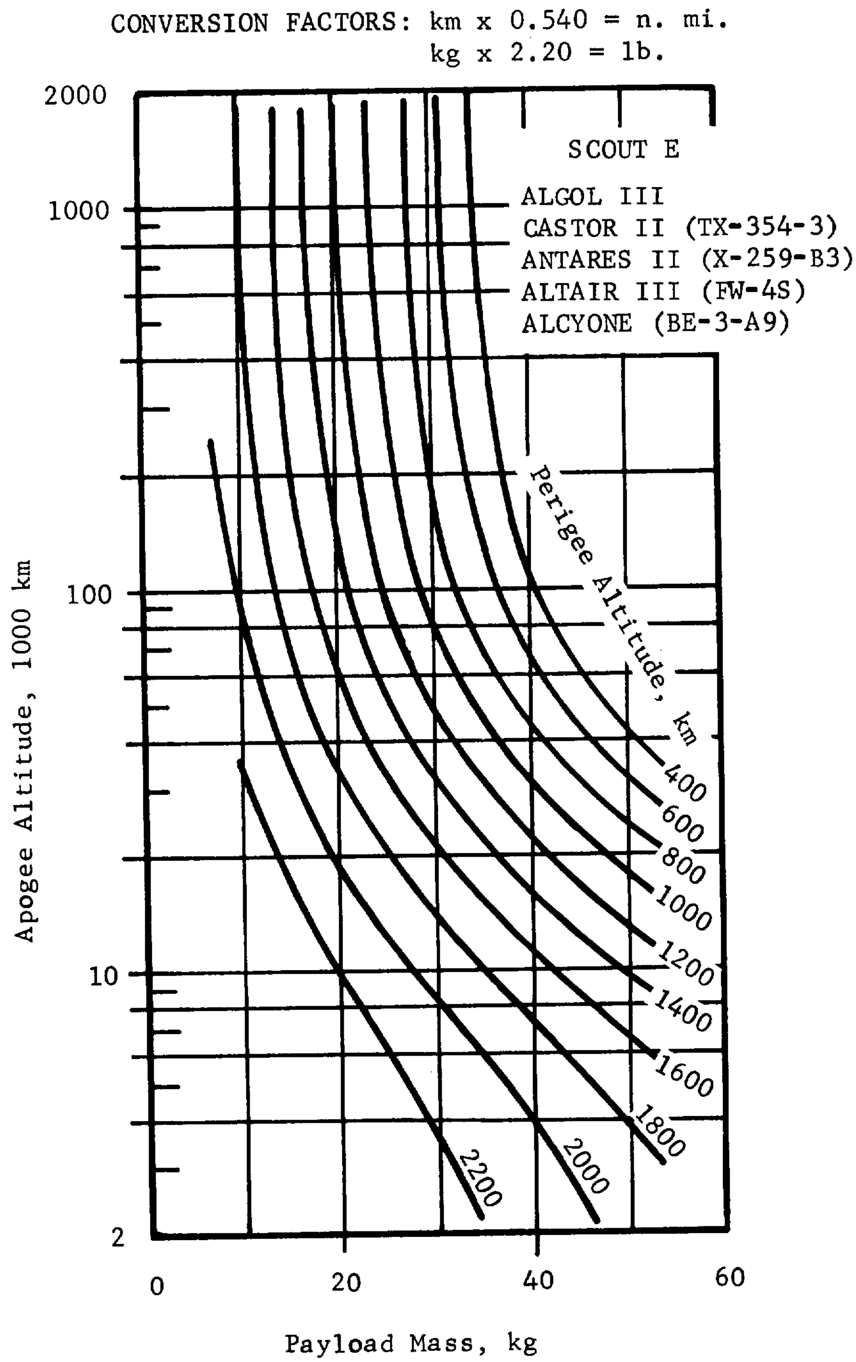


FIGURE 6-5. FIVE-STAGE SCOUT POLAR ORBITAL CAPABILITY, WTR

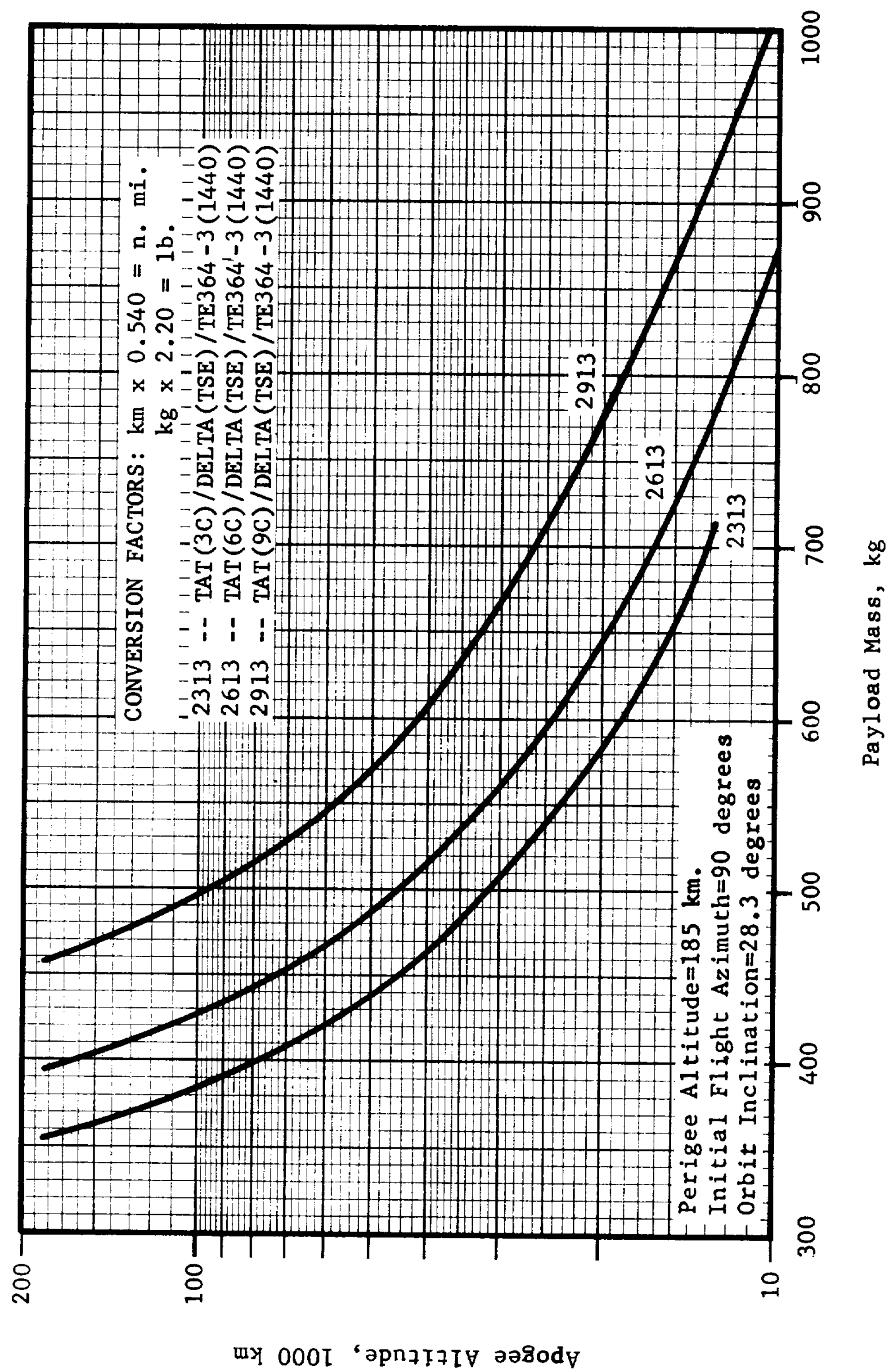


FIGURE 6-6. THREE-STAGE DELTA [TE364-3 (1440)] ORBITAL CAPABILITY-DUE EAST, ETR

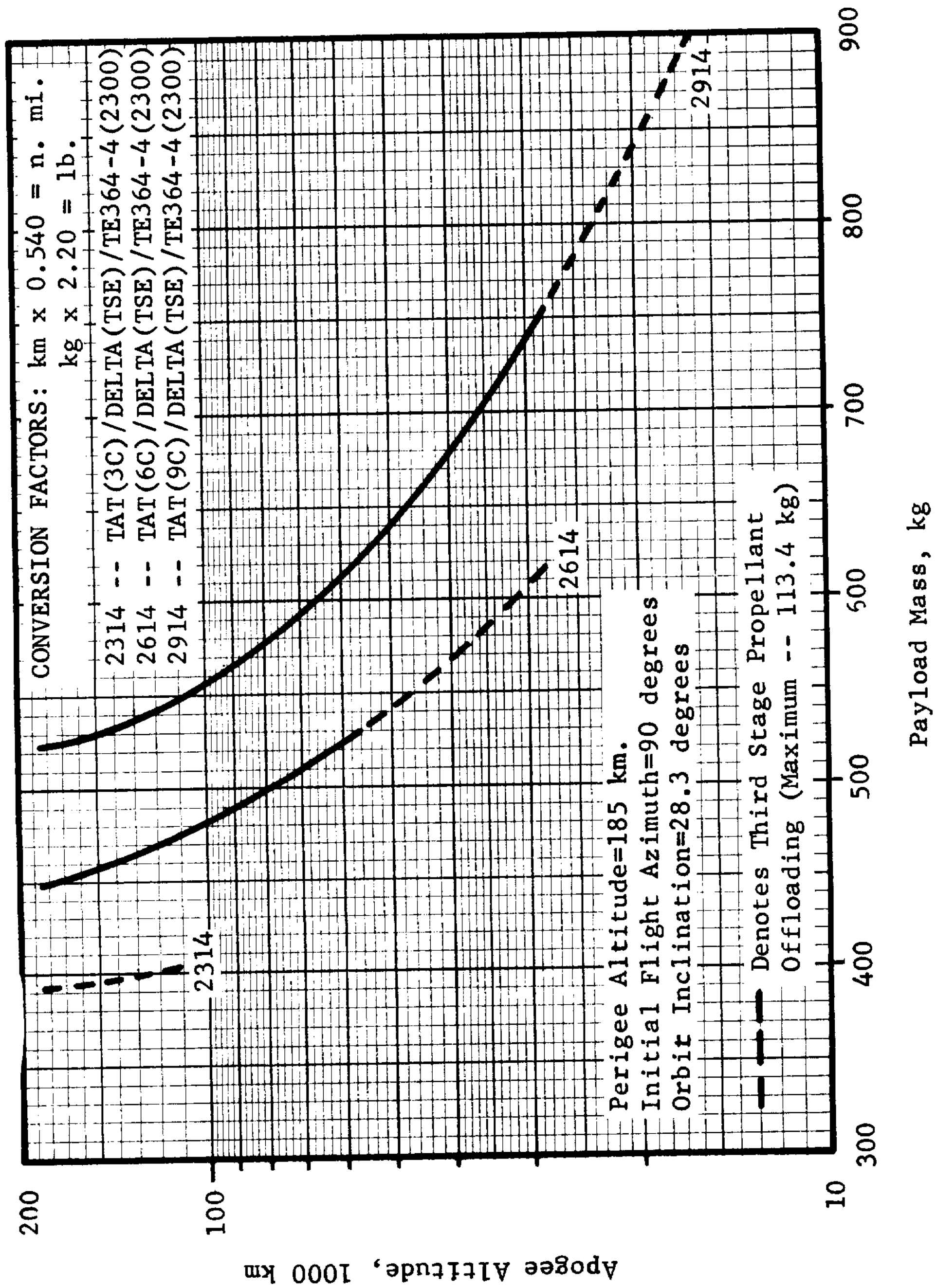


FIGURE 6-7. THREE-STAGE DELTA [TE364-4 (2300)] ORBITAL CAPABILITY-DUE EAST, ETR

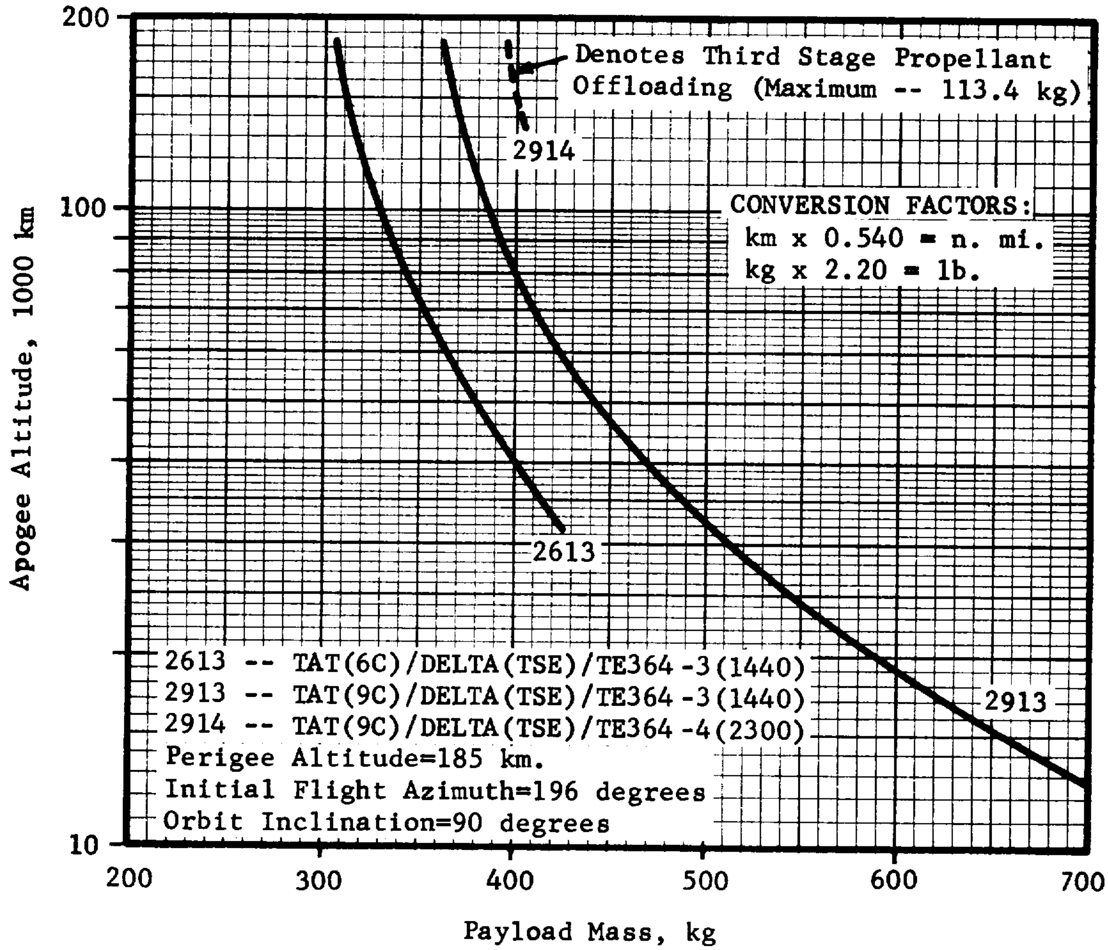


FIGURE 6-8. THREE-STAGE DELTA ORBITAL CAPABILITY, WTR

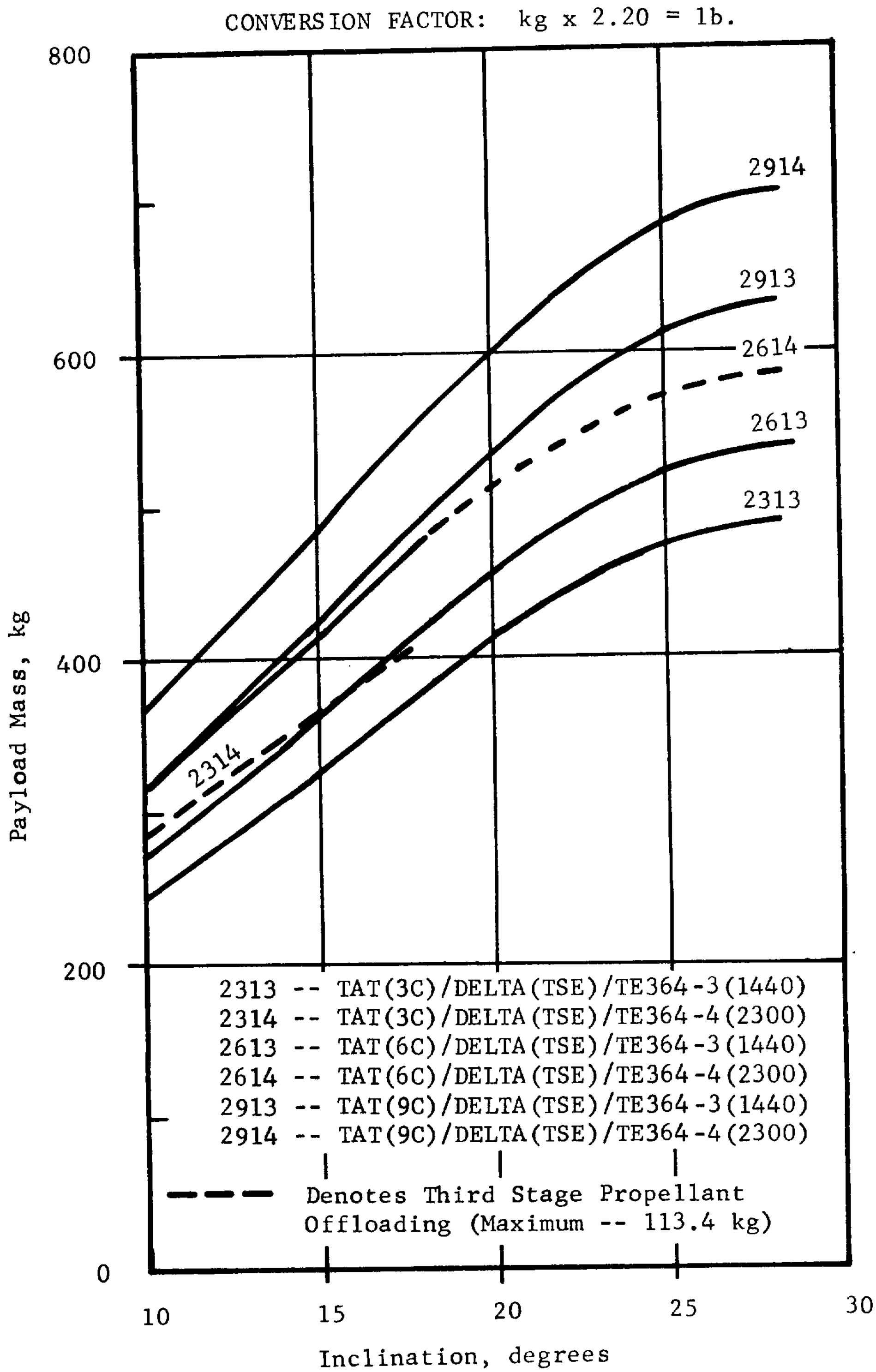


FIGURE 6-9. THREE-STAGE DELTA SYNCHRONOUS TRANSFER CAPABILITY, ETR

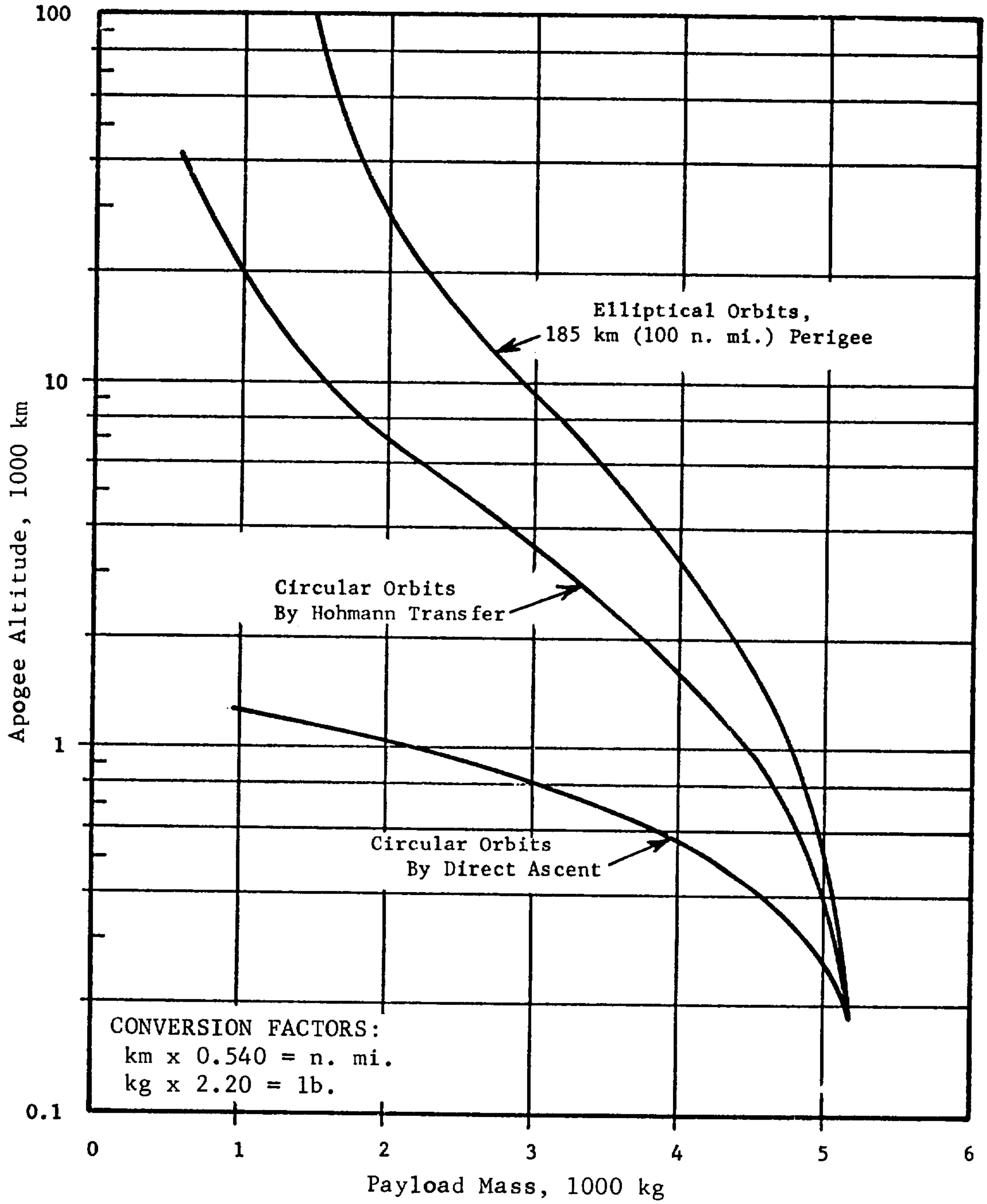


FIGURE 6-10. SLV3D/CENTAUR (SINGLE BURN) ORBITAL CAPABILITY-DUE EAST, ETR

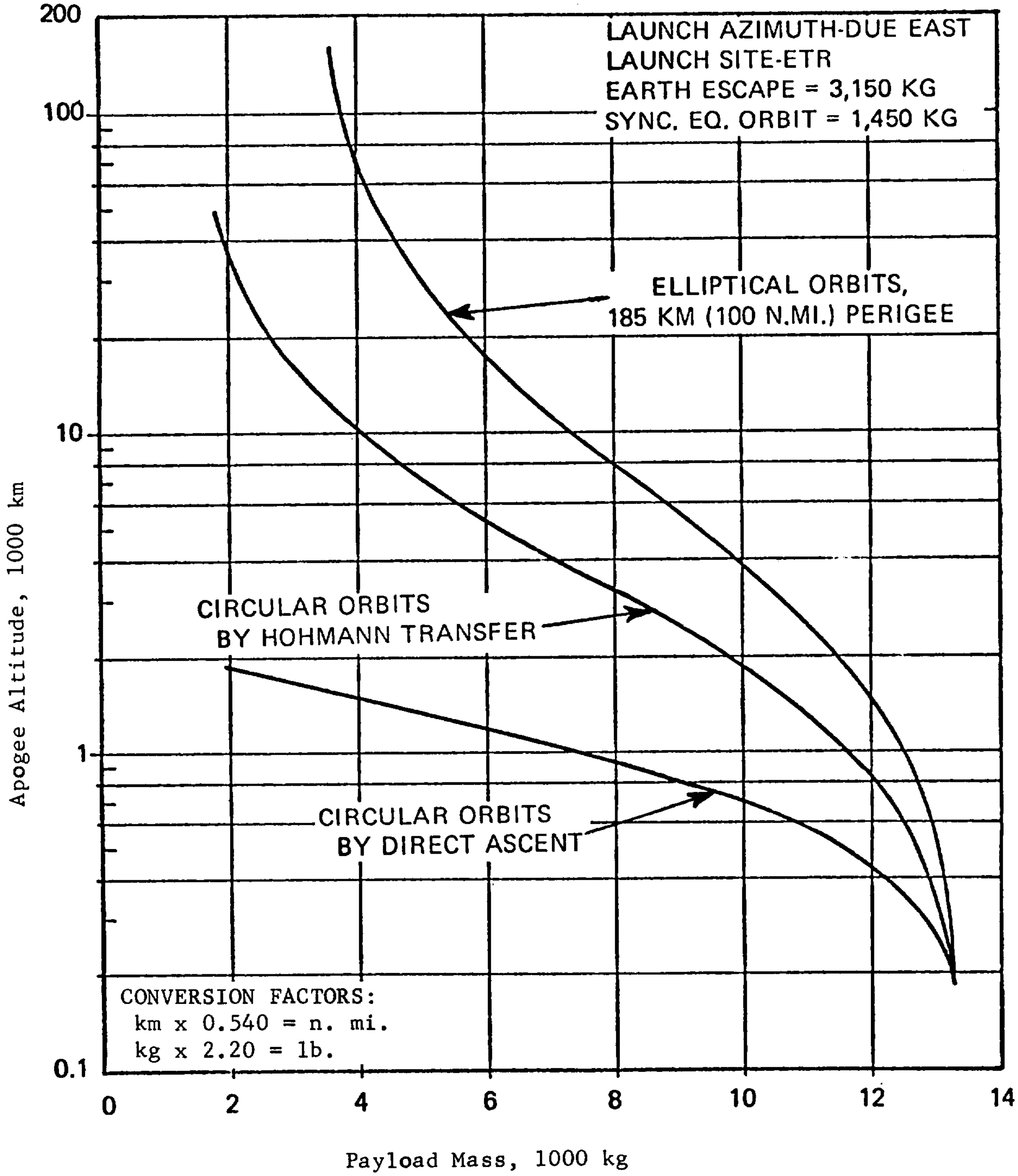


FIGURE 6-11. TITAN IIIC ORBITAL CAPABILITY-DUE EAST, ETR

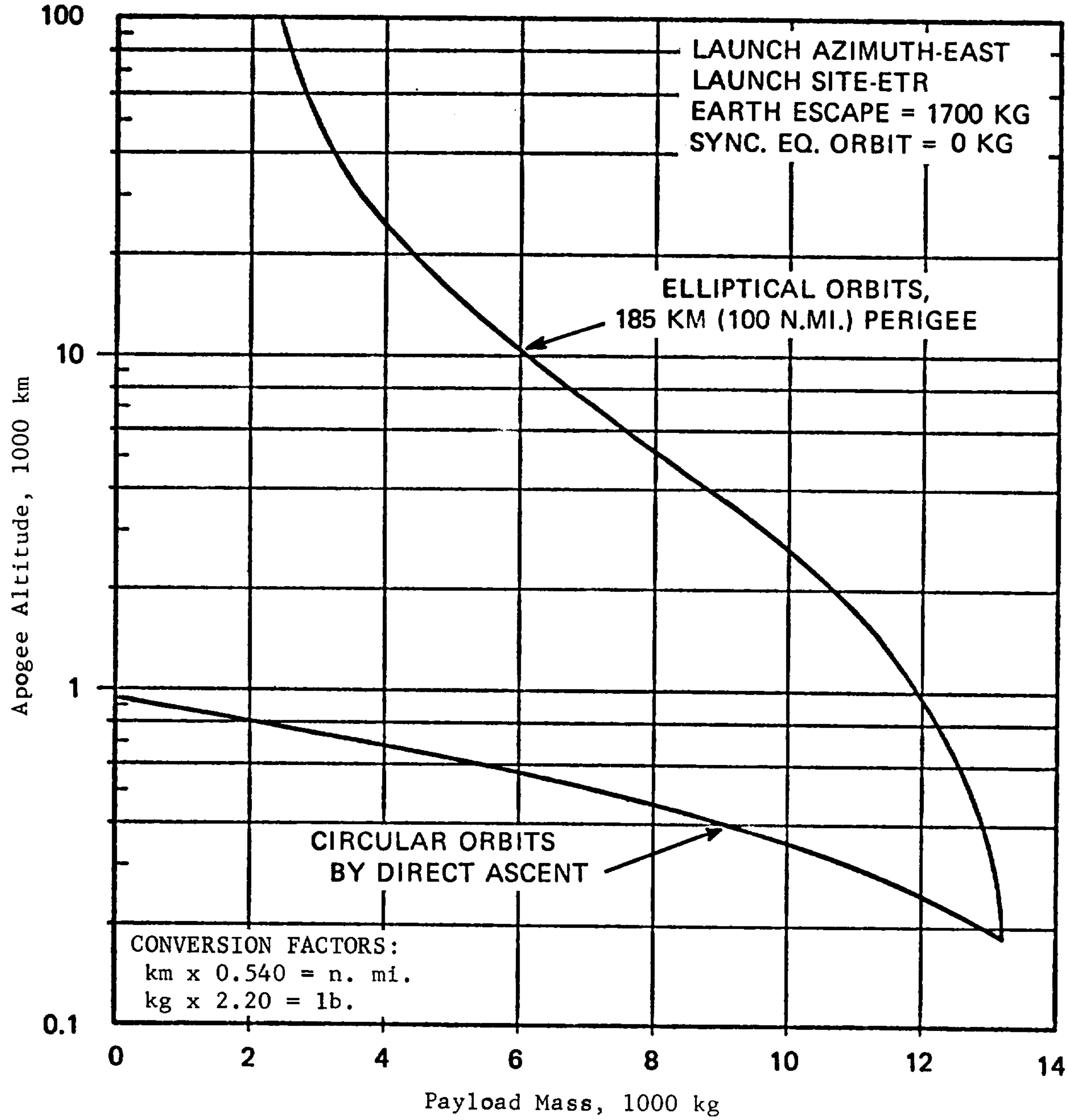


FIGURE 6-12. TITAN IIID ORBITAL CAPABILITY-DUE EAST, ETR

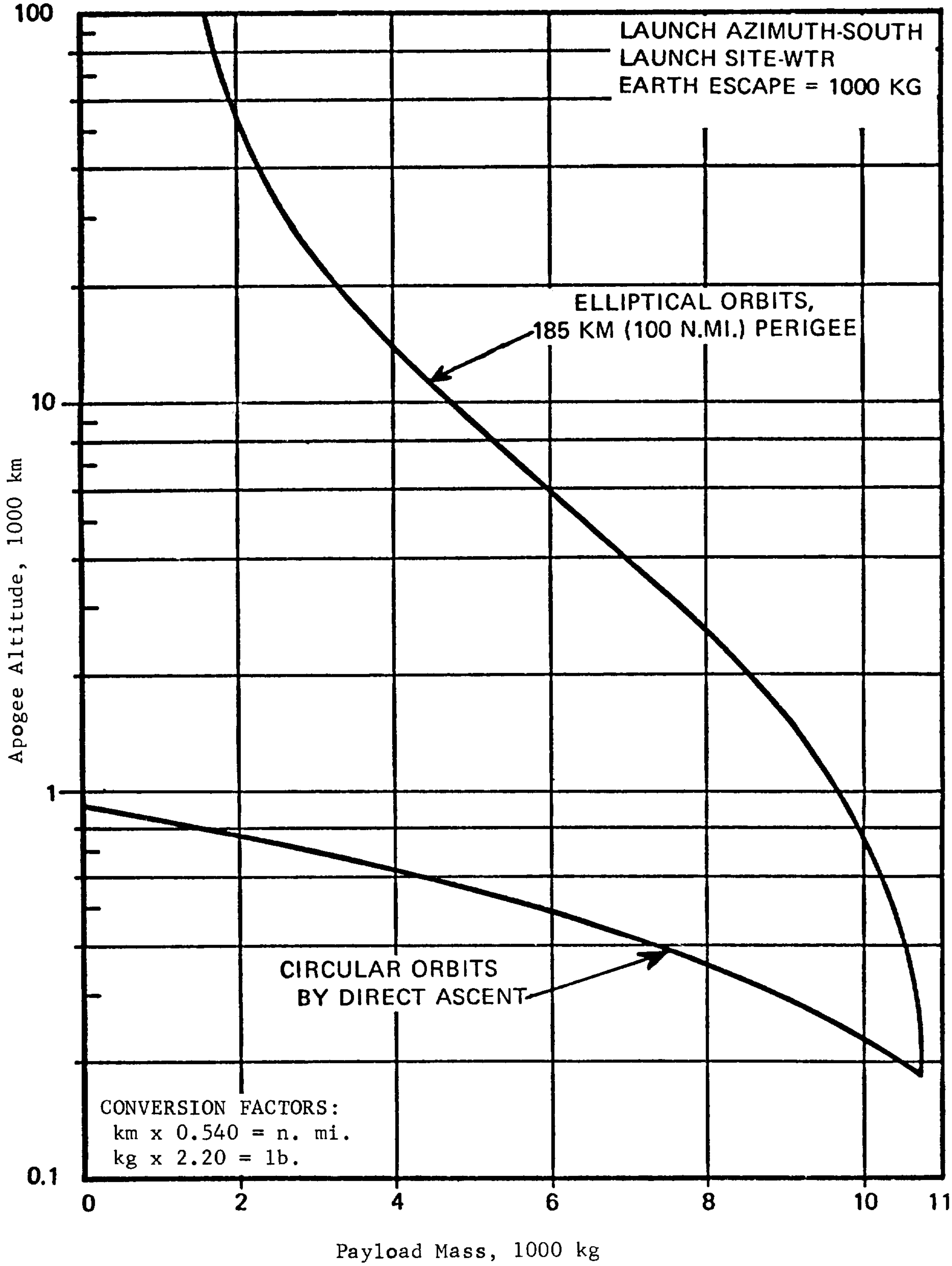


FIGURE 6-13. TITAN IIID POLAR ORBITAL CAPABILITY, WTR

CHAPTER 7: SPACE SHUTTLE PERFORMANCE

TABLE OF CONTENTS

<u>Figures-cont.</u> <u>Page</u>

CHAPTER 7: SPACE SHUTTLE PERFORMANCE

700 INTRODUCTION

1. This chapter presents data useful in preliminary planning estimates for missions using the space shuttle. The space shuttle is scheduled for operational use early in 1979 and should be considered in planning missions after that time.

2. Descriptive and performance data presented in this chapter are based on information in the "Space Shuttle Program Requirements Document, Level I" and the "Space Shuttle Baseline Payload Accommodations Document (Preliminary)" (References 28 and 29, Appendix B). These documents serve as the primary current sources of authoritative data on the space shuttle system.

701 SPACE SHUTTLE DESCRIPTION

1. The present baseline space shuttle system design consists of a reusable orbiter, external propellant tank, and dual solid rocket motors. The solid rocket motors may be recovered for refurbishment and reuse. The external propellant tank will be expendable.

2. The present space shuttle system definition allows for an effective all-azimuth launch capability by using launch sites at both the ETR and WTR. The space shuttle will thus satisfy the present and expected future ranges of launch-to-insertion azimuths. For the baseline design, liftoff is accomplished using the dual solid rocket motors and the orbiter main propulsion system to propel the space shuttle to the desired staging velocity and altitude. At staging, the solid rocket motors, having depleted their propellant, separate from the shuttle. The orbiter main propulsion system continues to operate, inserting the orbiter and the external propellant tank into a 93 x 185-km (50 x 100 n. mi.) reference injection orbit. After achieving this orbit, the external propellant tank is separated and subsequently deorbited for reentry into a selected disposal area.

3. For major orbital maneuvers starting from the reference injection orbit and for reentry, the orbiter will use an Orbital Maneuvering System (OMS). Integral OMS propellant tanks should

have sufficient capacity to provide a nominal ΔV capability of 305 m/sec (1000 fps). Up to 458 m/sec (1500 fps) of additional ΔV capability would be available by installing supplemental tankage in the orbiter payload bay. With the supplemental tankage, the OMS should have a total ΔV capability up to 763 m/sec (2500 fps). A Reaction Control System (RCS) would be provided for space attitude control and for terminal rendezvous maneuvers. An optional Air Breathing Engine System (ABES) will be available to provide cruise and loitering capability during atmospheric flight.

702 PAYLOAD ACCOMMODATIONS

1. A payload bay will be contained within the fuselage of the orbiter, providing a clear volume 4.57 m (15 ft) in diameter and 18.29 m (60 ft) long for the accommodation of payloads. With respect to shuttle missions, the term payload nominally incorporates all items of space hardware or cargo and associated payload support and ancillary equipment that will be transported to orbit within the payload bay. Some details about the accommodations of payloads aboard the orbiter remain to be determined. The fundamental criterion for payload accommodations is that of maintaining minimum interfaces between the orbiter and the payloads. Consequently, the payloads should be self-contained and self-sufficient to an extent consistent with the nominal accommodations to be provided by the orbiter.

2. Currently defined payload accommodations provided by the orbiter include the following items (References 28 and 29, Appendix B).

   a. Multiple sets of standardized attachment points would be provided along the payload bay to structurally support the payloads and to locate them within specified center-of-gravity limits. The attachment points would not obstruct the 4.57 x 18.29 m (15 x 60 ft) clear volume. Payload support equipment including structural pallets, shrouds, and special adapters which mount to the standardized attachment points will be provided by and charged to the payload.

   b. A standard deployment/retrieval mechanism will be available for handling payloads during orbital operations. The current baseline concept for this mechanism features a pair of manipulator arms attached to the forward bulkhead of the payload bay and controlled from an operations station within the orbiter cabin. Specialized erection, deployment, or retrieval mechanisms would be provided by and charged to the payload.

c. The orbiter avionics and computational facilities will be available on a time-sharing basis for payload support functions such as electrical power distribution and control, master caution and warning, navigational initialization, and limited functional end-to-end checkout of payload systems.

d. Up to 50 kwh of electrical energy (3 kw average power; 6 kw peak power) could be obtained from the orbiter electrical power system. Electrical energy in excess of this nominal allocation could be provided by the addition of extra fuel cell reactants or extra fuel cells chargeable to the payload.

e. Standard interfaces for payload fluid systems will be provided for emergency propellant dump and nonpropulsive fluid venting.

f. The nominal crew will be four persons – the orbiter commander, the pilot and two mission or payload specialists. Extra crew members or passengers and their provisions will be chargeable to the payload.

3. The payload supplier will be responsible for the following safety, reliability, and quality assurance activities:

a. Determination of the hazardous aspects of the payload and implementation of required safety measures.

b. Assurance of compatibility of the payload with the orbiter interfaces.

c. Identification of unresolved residual hazards and interface incompatibilities prior to approval of the payload.

d. The on-orbit functional reliability, quality, and safety of the payload.

4. In estimating the total or gross shuttle payload, it will be necessary to include not only the spacecraft or cargo (i. e., net payload) but also whatever structural pallets, adapters, shrouds, transfer stages, and ancillary equipment that may be required. Allowances for such items will vary over a wide range depending on the physical characteristics and functional requirements for

individual payloads. In some cases, such as that of a large payload approaching the limit of the orbiter payload or volume constraints, it may be desirable to design the structural pallet or support adapters and other payload support equipment as integral features of the payload. In other cases, it may be desirable to consider the use of a "common" set of payload support equipment such as proposed in References 30 and 31, Appendix B. Preliminary guidelines for estimating allowances for payload support equipment, based on these references, are given in the following paragraphs.

5. Table 7-1 presents a summary of preliminary data pertaining primarily to the accommodation of automated spacecraft, either with or without a transfer stage, and unmanned experiments. This common set of payload support hardware should accommodate single or multiple (up to five) payloads with a minimum of customized or mission-peculiar equipment. The items of equipment (see Table 7-1 for descriptions) might be used singly or in various combinations depending upon individual payload requirements.

6. Table 7-2 presents a summary of preliminary data pertaining primarily to sortie-type (manned) missions. The data are for habitable modules and include allowances for a life support system, extra crew members, and extra crew provisions over and above that nominally provided by the orbiter.

7. Using the data provided in Table 7-1, Table 7-2, or using other appropriate data, the payload planner can estimate the total allocation for the payload and payload support equipment that comprise the gross shuttle payload. Having defined the mission in terms of gross shuttle payload and orbital specification or space destination, it is then necessary to determine whether the mission can be accomplished by direct delivery using the shuttle alone, or whether some type of transfer stage would be required to complete the mission starting from an appropriate shuttle parking orbit. To facilitate this analysis, the shuttle performance data in this chapter are presented in two parts. First, information for the planning of direct delivery missions are presented in paragraph 703. Second, data for the planning of missions involving placement of payloads together with transfer stages in low Earth parking orbits for subsequent transfer to other orbits are presented in paragraph 704.

TABLE 7-1. ESTIMATES FOR PAYLOAD SUPPORT AND DELIVERY EQUIPMENT

Item	Mass, kg (lb)	Dimensions: Length, m (ft)	Dimensions: Diameter, m (ft)	Functional Description
General Services Unit	1, 257 (2, 769)	2. 96 (9. 71)	4. 1 (13. 4)	Cylindrical module for mounting payload ancillary equipment and sensors.
Basic Structure and Adapters	942 (2, 075)	--	--	
Basic Subsystems	315 (694)	--	--	
Strongback	1, 262 (2, 782)	11. 61 (38. 1)	4. 57 (15. 0)	Structural pallet for mounting payload cargo and/or ancillary equipment. Serves as the lower half of a payload shroud. Incorporates structural adapters for mating with the orbiter attachment points.
Strongback Extension	318 (700)	3. 25 (10. 7)	4. 57 (15. 0	
Shroud (Including End Cap)	761 (1, 678)	11. 61 (38. 1)	4. 57 (15. 0)	Together with the strongback, the shroud provides an enclosure for payload environmental control.
Shroud Extension	213 (470)	3. 25 (10. 7)	4. 57 (15. 0)	
Umbilical Beam	310 (684)	8. 51 (27. 9)	--	A cantilever structure attachable to the General Services Unit for supporting spacecraft payloads and spacecraft/transfer stage payloads. The umbilical beam could be made to serve as an erection/deployment mechanism.

CAUTIONARY NOTE: These data are based on preliminary contractor studies and do not necessarily reflect official policies of the NASA Space Shuttle Program. The data are presented as guidelines for preliminary payload planning. See Reference 30, Appendix B.

TABLE 7-2. ESTIMATES FOR SORTIE MISSION EQUIPMENT

Item	Mass, kg (lb)	Dimensions Length, m (ft)	Dimensions Diameter, m (ft)	Functional Description
REQUIRED ITEMS				
Sortie Module	4,634 (10,216)	5.49 (18.0)	4.27 (14.0)	Habitable (pressurized) module for supporting on-orbit experiment.
Basic Structure	2,488 (5,486)	--	--	
Subsystems, Provisions	2,146 (4,730)	--	--	
Pallet	1,000 (2,204)	7.87 (25.8)	4.06 (13.3)	Structural pallet (unpressurized) for mounting payloads and ancillary equipment.
OPTIONAL ITEMS				
Support Module	6,809 (15,012)	5.49 (18.0)	4.27 (14.0)	Habitable (pressurized) module to accommodate additional mission/payload specialists or passengers over and above the nominal allocations for two mission/payload specialists provided by the shuttle. Used in conjunction with Sortie Module and, if necessary, a pallet.
Basic Structure	2,412 (5,318)	--	--	
Subsystems	2,622 (5,781)	--	--	
Extra Crew and Provisions	587 (1,295)	--	--	
Miscellaneous	1,188 (2,618)	--	--	
Payload Module				
18' Version	2,630 (5,798)	5.49 (18.0)	4.27 (14.0)	Pressurized module for special experiments requiring direct access by crew. Used in conjunction with Sortie Module and other items of equipment as required.
Basic Structure	2,238 (4,934)	--	--	
Basic Subsystems	392 (864)	--	--	
32' Version	3,966 (8,744)	9.75 (32.0)	4.27 (14.0)	
Basic Structure	3,536 (7.795)	--	--	
Basic Subsystems	430 (949)	--	--	

CAUTIONARY NOTE: These data are based on preliminary contractor studies and do not necessarily reflect official policies of NASA Space Shuttle Program. The data are presented as guidelines for preliminary payload planning. See Reference 31, Appendix B.

703 SHUTTLE DIRECT DELIVERY CAPABILITIES

1. General

The space shuttle concept defines a capability for versatile operations, including the direct delivery (no transfer stage) of payloads to low Earth orbits. From the two space shuttle launch sites that have been selected, ETR and WTR, orbit inclinations that can be attained by direct shuttle flights range from 28.5 degrees to more than 120 degrees except when range safety imposes launch azimuth constraints. Figures 4-1 and 4-2 should be consulted for the presently accepted range safety limits on launch azimuth. Users with mission requirements within these general limits should consider the direct delivery mode. Shuttle gross payload capabilities for circular orbits are described in paragraph 703.2. Paragraph 703.3 describes elliptical orbit capabilities of the shuttle.

2. Circular Orbit Capabilities

a. Figure 7-1 shows shuttle gross payload capabilities as a function of circular orbit altitude for different inclinations. Two payload scales are shown representing shuttle performance with and without the Air Breathing Engine System. For this performance map, it was assumed that the orbiter would always be injected into the 93 x 185 km (50 x 100 n. mi.) reference injection orbit. All subsequent maneuvers would be performed using the OMS. Payload can be traded directly for OMS propellant until the OMS propellant tanks are full. It was assumed that the entire payload would be carried throughout all maneuvers. This ensures that the orbiter will be able to deorbit in the event that the payload could not be deployed or that another payload was retrieved for return to Earth. A constant OMS ΔV reserve of 15 m/sec (50 fps) was assumed. No allowance is included for rendezvous. If rendezvous is required, an extra OMS ΔV of 37 m/sec (120 fps) must be budgeted. This would reduce the circular orbit altitude that could be reached with any payload by 46.3 km (25 n. mi.).

b. Figure 7-2 gives gross shuttle payload as a function of orbit inclination for different circular orbit altitudes. This figure is a cross-plot of Figure 7-1 and the assumptions and qualifications discussed in subparagraph 703.2a apply.

c. Example problem: Deliver a 3,000-kg payload and Sortie Module to a 300-km, 90° polar circular orbit with the ABES installed in the shuttle. From Table 7-2, the mass of the payload pallet and sortie module are found to be 1,000 and 4,634 kg, respectively. The total gross payload would be 1,000 + 4,623 + 3,000 = 8,623 kg. Figure 7-2 shows that 9,600 kg is the gross payload with the ABES. Therefore, the mission can be performed.

d. Example problem: Deliver 10,000 kg of cargo to a 700-km, 28.5° circular orbit with the ABES installed in the shuttle. Assume that a strongback and a strongback extension with a mass of 1,580 kg (Table 7-1) would be required. The gross payload is then 10,000 + 1,580 kg = 11,580 kg. Figure 7-2 shows that the shuttle can deliver a gross payload of 11,800 kg and, therefore, the mission is possible.

3. Elliptical Orbit Capabilities

a. Elliptical orbits and circular orbits for the shuttle have no simple one-to-one correspondence as far as performance is concerned. This is because the entry ΔV required for a highly elliptical orbit may vary from a few hundred m/sec to achieve entry at perigee to as much as several thousand m/sec if an intermediate phasing orbit is required. Figures 7-3 and 7-4 show the payload that can be carried by the shuttle for a 185 km (100 n. mi.) perigee as a function of apogee altitude. Figure 7-3 is for orbits of 28.5 to approximately 58-degree inclinations and Figure 7-4 is for polar orbits of 90-degree inclination. In computing the curves on Figures 7-3 and 7-4 it was assumed that the shuttle orbiter would first be inserted into a 93 x 185 km (50 x 100 n. mi.) reference injection orbit. Using the OMS, the orbit would then be raised to a 185-km (100 n. mi.) circular orbit before insertion into the elliptical orbit.

b. The higher performance curve on each graph represents the situation in which the landing site location is compatible with perigee location so that a minimum retroburn would be required. This is an ideal situation and is representative of the maximum altitudes that can be reached by the shuttle, consistent with the nominal design constraints for the thermal protection system. The lower performance curve represents the case in which insertion would again be made into an elliptical orbit, but the desired landing site location is such that the elliptical orbit would have to be reduced to a

185-km (100 n. mi.) circular phasing orbit before retrofiring. The realistic limits of payload and operating altitude would lie between the two curves and each mission would have to be examined individually to determine maximum performance capabilities.

c. With the shuttle launched into a highly elliptical orbit, a payload could be deployed at apogee altitude and placed into a circular orbit with a single propulsive burn of an apogee kick motor. This maneuver is described in paragraph 704.

d. Direct reentry from the higher orbits available to the shuttle can result in reentry velocities as much as 610 m/sec (2,000 ft/sec) higher than the nominal design conditions. Such reentries would have various additional reentry angle and range constraints imposed in order to assure safe return. These constraints would depend upon the final design and are not yet well defined. In general, missions requiring direct shuttle reentry from the higher altitudes should be planned in coordination with the persons listed in the Preface to assure that such reentry constraints are not violated.

704 SHUTTLE PLUS TRANSFER STAGE CAPABILITIES

1. General

a. Missions to orbit altitudes higher than the direct shuttle capability shown in Figures 7-1 and 7-2 and those missions outside the range of shuttle orbit inclinations (refer to paragraph 703. 1) would require additional propulsion stages. These stages would be transported within the shuttle cargo bay to a low Earth parking orbit where they would be deployed to complete the required mission. For inclinations less than 28.5 degrees, the shuttle could be placed in a parking orbit at 28.5 degrees and the transfer stage would perform the required plane change.

b. Table 7-3 presents masses, dimensions, and performance characteristics of upper stages using liquid propellants suitable for use as shuttle transfer stages. These include the existing Delta, Agena, Transtage, and Centaur stages. All except the Centaur use space-storable propellants. Similar data for solid propellant motors are presented in Chapter 8.

TABLE 7-3. CHARACTERISTICS OF REPRESENTATIVE SHUTTLE TRANSFER STAGES

Designation	Length Diameter, m (ft)	Vacuum Specific Impulse, m/s (sec)	Stage Mass (Ignition)(a), kg (lb)	Stage Mass (Burnout)(a), kg (lb)	Adapter Mass, kg (lb)
Solid-Propellant Motors – See Chapter 8, Table 8-1					
Liquid-Propellant Stages (Storable)					
Delta	5.88/1.46 (19.3/4.8)	2,980 (304)	5,477 (12,074)	859 (1,894)	23-36 (50-80)
Agena	6.17/1.52 (20.2/5.0)	2,850 (291)	6,772 (14,930)	657 (1,448)	27-45 (60-100)
Transtage	4.50/3.05 (14.75/10.0)	2,960 (302)	12,464 (27,477)	1,844 (4,065)	104 (230)
Liquid-Propellant Stages (Cryogenic)					
Centaur	9.14/3.05 (30.0/10.0)	4,292 (438)	15,620 (34,436)	1,984 (4,374)	53 (116)

(a) Stage mass excludes nominal adapter mass indicated in the last column.

c. Generalized performance data for restartable transfer stages are presented in paragraph 704.2. Corresponding data for nonrestartable transfer stages are discussed in paragraph 704.3.

2. Shuttle Plus Restartable Transfer Stage Capabilities

a. Figures 7-5 through 7-10 show net spacecraft payload capabilities as a function of characteristic velocity (V_C) for the shuttle together with various liquid propellant transfer stages. All of these stages are restartable and in most cases can be programmed to deliver the proper sequence of velocity impulses to accomplish Earth-orbital and Earth-escape missions. The payload values plotted along the ordinate of these figures represent actual or net spacecraft payload. Adjustments have been made for the masses of the shuttle interface equipment (consisting of a strongback, general service unit, umbilical, and shroud with a total mass of 3,590 kg), transfer stage, and transfer-stage/spacecraft adapter. If the total adjustments plus the net payload exceed the gross shuttle payload capability, it is necessary to begin off-loading the transfer stage propellants to achieve higher net payload values. This results in a relatively pronounced break in the payload-V_C curves at the point where off-loading begins.

b. Normally, payload-V_C curves such as those in Figures 7-5 and 7-6 are valid for a specific value of inclination and a velocity correction factor must be applied in order to account for launches to other inclinations. However, as can be seen in Figures 7-1 and 7-2 the shuttle payload capability for inclinations between 28.5 and 58 degrees at the 185-km (100 n. mi.) reference altitude is constrained by the structural limit and not by performance. Because of its excess performance capability, the shuttle can deliver the extra ΔV necessary to reach orbit inclinations up to approximately 58 degrees at the 185-km reference orbit without incurring any payload penalty. For this reason, Figures 7-5 and 7-6 are valid for inclinations between approximately 28.5 and 58 degrees.

c. Figures 7-7 to 7-10 show performance capabilities of various shuttle transfer stages for near-polar orbits. For these higher orbital inclinations, the shuttle payload capability is constrained by performance. Thus, the maximum gross payload for a 185-km (100 n. mi.) circular orbit will vary as a function of inclination as can be seen in Figure 7.2.

Correspondingly, the point at which the off-loading of transfer-stage propellants must begin also varies as a function of inclination. The curves in Figures 7-7 to 7-10 depict the performance with and without the Air Breathing Engine System aboard the Orbiter.

d. As an indication of the mission capabilities represented by characteristic velocities, an equatorial synchronous orbit requires a total characteristic velocity of about 12.1 km/sec (39,600 ft/sec) via a 185-km (100 n. mi.) circular parking orbit at a 28.5-degree inclination. Characteristic velocity requirements for Mars and Venus encounters are normally between 11.3 and 12.0 km/sec (37,200 and 39,500 ft/sec). Translunar injection requires 11.0 to 11.3 km/sec (36,000 to 37,000 ft/sec). Outer planetary missions range from 14.6 km/sec (48,000 ft/sec) upwards. Characteristic velocities for specific missions can be determined from the information in Chapters 2 and 3.

e. For the case of small payloads destined for high energy trajectories, the performance capability can be significantly improved by the addition of a small velocity package. A curve representative of the increased performance capability available through the use of such velocity packages – e.g., Burner II (2300) – is shown for the shuttle/Centaur third stage in Figures 7-6 and 7-7. The performance of other such velocity packages can be derived easily using a three-step procedure outline in paragraph 801. The following example illustrates the procedure.

f. Example Problem: An asteroid probe requires a characteristic mission velocity of 13.1 km/sec. The payload is 700 kg. Can the shuttle be used to perform this mission? Solution: Figure 7-5 (or 7-6) shows that the mission requirements fall slightly above the performance capability of the Shuttle/Agena and the Shuttle/Transtage. It would be worthwhile to consider using a velocity package rather than going to the larger Centaur third stage. From Figure 8-1, for a payload of 700 kg, it is found that a Burner II (1440) would give a ΔV of 1.53 km/sec. From Table 8-1 the total mass of the BII (1440) and its payload adapter is 805 + 9 + 0.1 (700-250) = 859 kg. If the velocity package is included as part of the spacecraft, the "net shuttle payload" would be 859 + 700 = 1,559 kg and, for that payload, the Shuttle/Agena would give a V_C of 11.55 km/sec. The velocity package would give an additional 1.53 km/sec making the total characteristic velocity 11.60 + 1.53 = 13.13 km/sec. Therefore, the Shuttle/Agena plus a BII (1440) could perform the mission.

3. Shuttle Plus Nonrestartable Transfer Stage Capabilities

a. Figure 7-11 shows net payload as a function of characteristic velocity for the shuttle with several representative versions of Burner II stages. This figure is included to indicate the general range of performance capabilities for the shuttle with solid propellant velocity packages.

b. Certain precautions are necessary for advance planning of shuttle missions using solid propellant third stages. In general, it is necessary to estimate all of the incremental velocity impulses (ΔV's) required to perform a given mission and then to select a velocity package which, in combination with the shuttle, would not only give the proper characteristic velocity (V_C) but, also, the proper sequence of velocity impulses (ΔV's). Since all velocity packages presently available incorporate one or more single-burn motors, it is necessary to ensure that the ΔV's per motor (i.e., per stage) are equal to or greater than the corresponding ΔV's required to perform the mission.

c. The use of velocity packages might also be advantageous for missions in which the shuttle OMS propulsion system would be used to establish an elliptical parking orbit with an apogee altitude corresponding to the final desired apogee value. Then, a velocity package could be used to provide the single ΔV impulse necessary for establishing a new perigee altitude or for circularization. In this case, an appropriate procedure discussed in paragraphs 302 through 304 would be used to find the magnitude of the velocity impulse required to perform the maneuver. Then, these steps should be followed:

(1) Select a velocity package from Figure 8-1 which gives the required ΔV for the proposed spacecraft payload. For a given payload, it is unlikely that a velocity package would be available that gives exactly the required ΔV's. If this is the case, a velocity package should be selected which gives ΔV's slightly greater than those desired. Several techniques are available to compensate for the extra performance.

(2) Add the mass of the proposed payload to the mass listed in Table 8-1 for the velocity package and its adapter. If the payload exceeds the limit for which the velocity package structure is designed, it will be necessary to account for the additional mass of a strengthened structure according to procedures described in the footnotes on Table 8-1.

(3) Assuming that the total mass of the velocity package and the proposed payload represents the "net" payload of the shuttle, find the required shuttle ancillary equipment mass using Table 7-1 or Table 7-2, if appropriate. Sum these masses to obtain the gross shuttle payload.

(4) Compare the gross payload found in Step (3) with the shuttle performance capabilities for elliptical orbits shown on Figure 7-3 or 7-4 to ensure that the shuttle performance capabilities are not exceeded.

d. For Earth orbital missions requiring more than one velocity impulse it is necessary to ensure that the velocity package provides a sequence of velocity impulses (ΔV's) corresponding to the mission requirements. Again, mission requirements can be determined using procedures described in paragraphs 302 through 304. Figure 8-2 gives incremental ΔV's per stage (i.e., per motor) for several two-stage velocity packages including those shown in Figure 7-11. The data in Figure 8-2 can be used to select a multistage velocity package that gives the proper sequence of velocity impulses to perform a given mission. The procedure would be the same as in the four steps listed previously except that in Step (1) Figure 8-2 would be used to select a velocity package which would give the proper sequence of velocity impulses and in Step (4) Figure 7-1 or 7-2 would be used if the shuttle-parking orbit is circular rather than elliptical. The following example illustrates this procedure.

e. Example Problem: Launch a spacecraft payload of 900 kg into a circular Sun-synchronous orbit (100.0 degrees) at 1,300-km altitude. According to Figures 7-1 and 7-2 this mission is beyond the shuttle-only capabilities so that a shuttle transfer stage would be required. Consider two alternative solutions: one in which the shuttle would be launched into an elliptical parking orbit and another in which the shuttle parking orbit is circular.

(1) Elliptical Parking Orbit: A 185 x 1300-km shuttle parking orbit at a required inclination of 100 degrees (see Figure 3-6) is assumed. From Figure 3-4c, the ΔV required for circularization is about 287 m/sec. Using Figure 8-1, it is found that a velocity package as small as the OV1 (FW4) would provide adequate performance. The mass of that velocity package is found from Table 8-1 to be 390 kg which, together with the payload, gives a net shuttle payload of 1,290 kg. From Table 7-1, the desired payload interface with the shuttle can be chosen. For this case, the strongback, general service unit, shroud, and umbilical are chosen, having a total mass of 3,590 kg. The gross shuttle payload becomes 4,800 kg. On the basis of Figure 7-4, it can be estimated that the gross shuttle payload capability without the ABES is sufficient, although supplemental tankage may be required. Thus, the mission can be performed without the ABES using the OV1(FW4) velocity package.

(2) Circular Parking Orbit: A 185-km circular parking orbit at the required inclination of 100 degrees is assumed. From Figure 3-4c is found that the velocity impulses required for transfer and circularization are approximately 300 and 287 m/sec, respectively. For a payload of 900 kg, the Burner IIA(1440/524) provides more than adequate performance, as can be determined using Figure 8-2. The total mass of the Burner IIA(1440/524) and its payload adapter is 1,120 + 9.0 + 0.125 (900-250) = 1,210 kg as indicated in Table 8-1. The "net" shuttle payload is therefore 900 + 1,210 = 2,110 kg. From Table 7-1 the desired payload interface equipment can be chosen. For this case, the strongback, general service unit, shroud, and umbilical are chosen, having a total mass of 3,590 kg. Thus, the gross shuttle payload would be 2,110 + 3,590 kg = 5,700 kg. This is well within the shuttle performance capabilities shown on Figures 7-1 and 7-2, both with and without the ABES. Therefore, the mission could be performed using the BIIA(1440/524) velocity package.

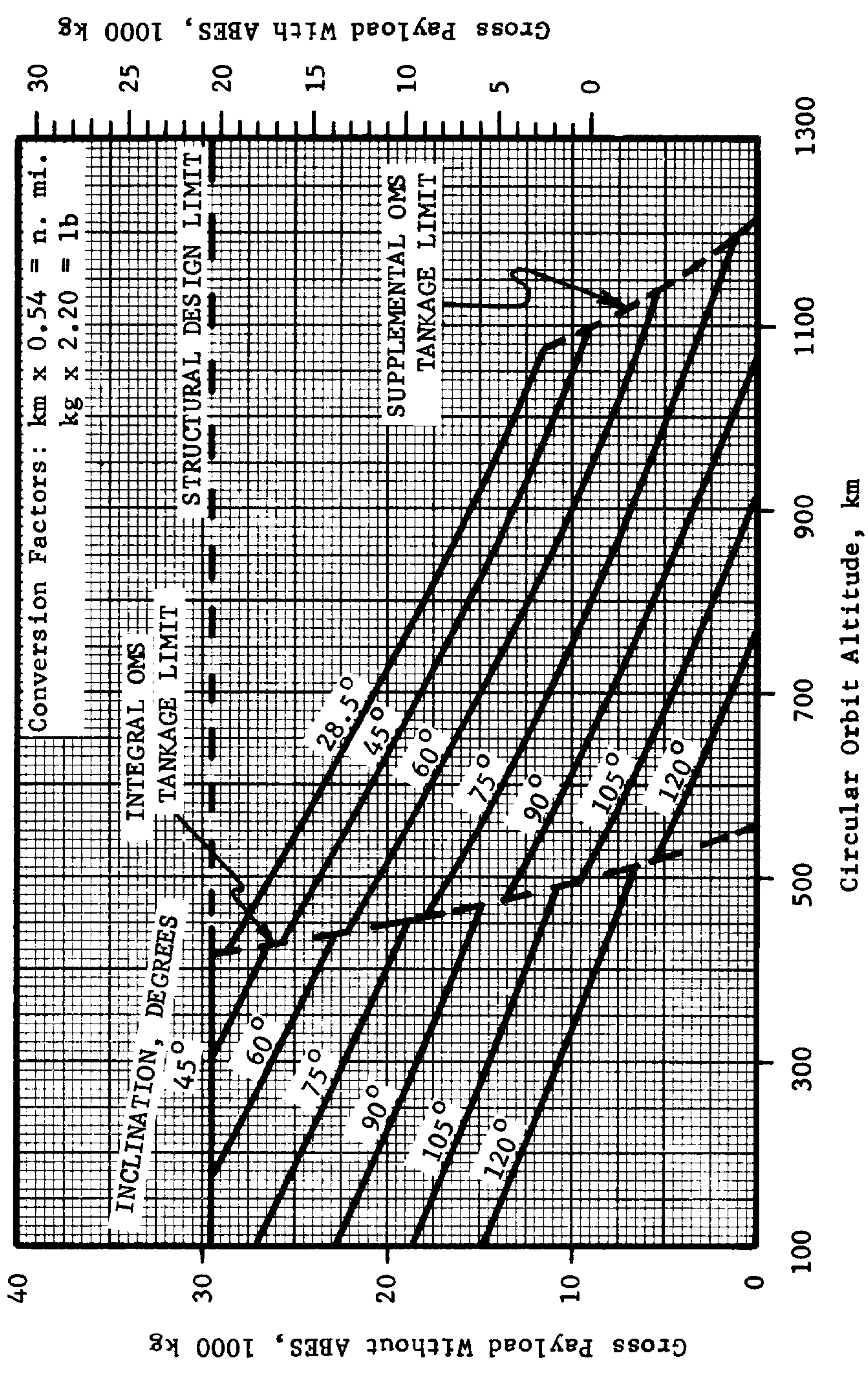


FIGURE 7-1. SPACE SHUTTLE GROSS PAYLOAD TO CIRCULAR ORBIT ALTITUDE FOR VARIOUS ORBITAL INCLINATIONS

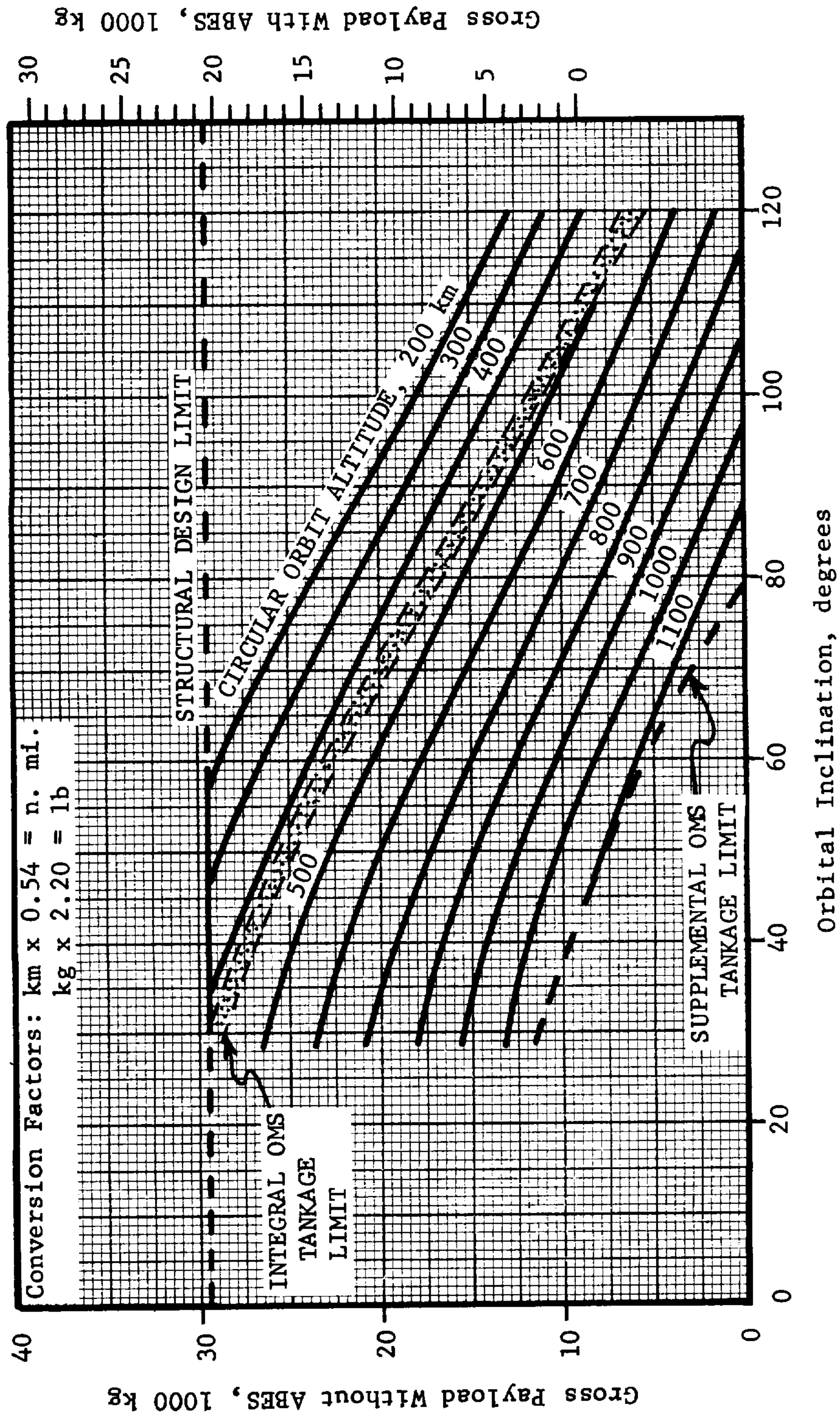


FIGURE 7-2. SPACE SHUTTLE GROSS PAYLOAD TO VARIOUS ORBITAL INCLINATIONS

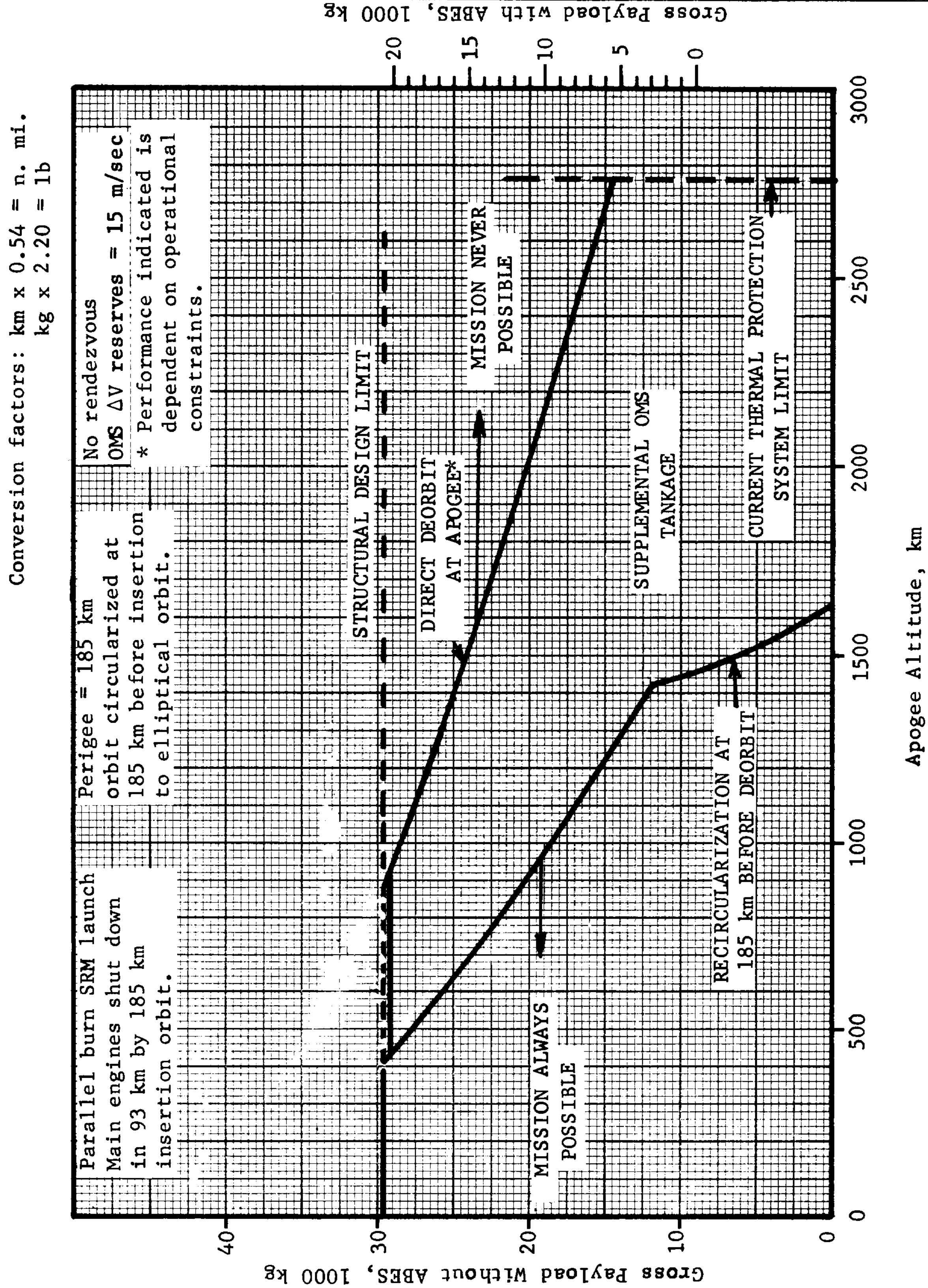


FIGURE 7-3. SPACE SHUTTLE GROSS PAYLOAD TO ELLIPTICAL ORBIT FOR ORBITAL INCLINATIONS BETWEEN 28.5 AND 58 DEGREES

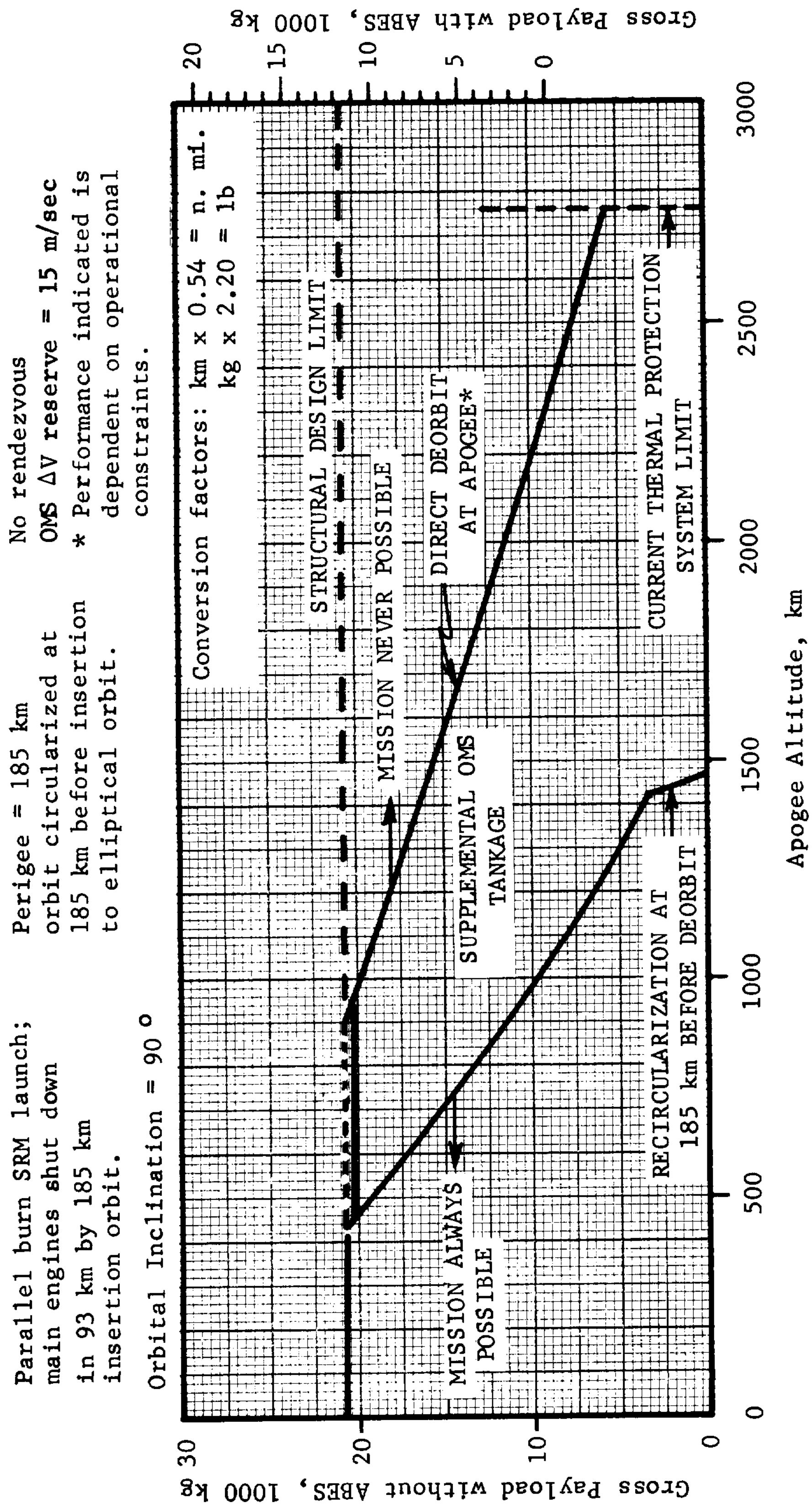


FIGURE 7-4. SPACE SHUTTLE GROSS PAYLOAD TO ELLIPTICAL ORBIT AT AN ORBITAL INCLINATION OF 90 DEGREES

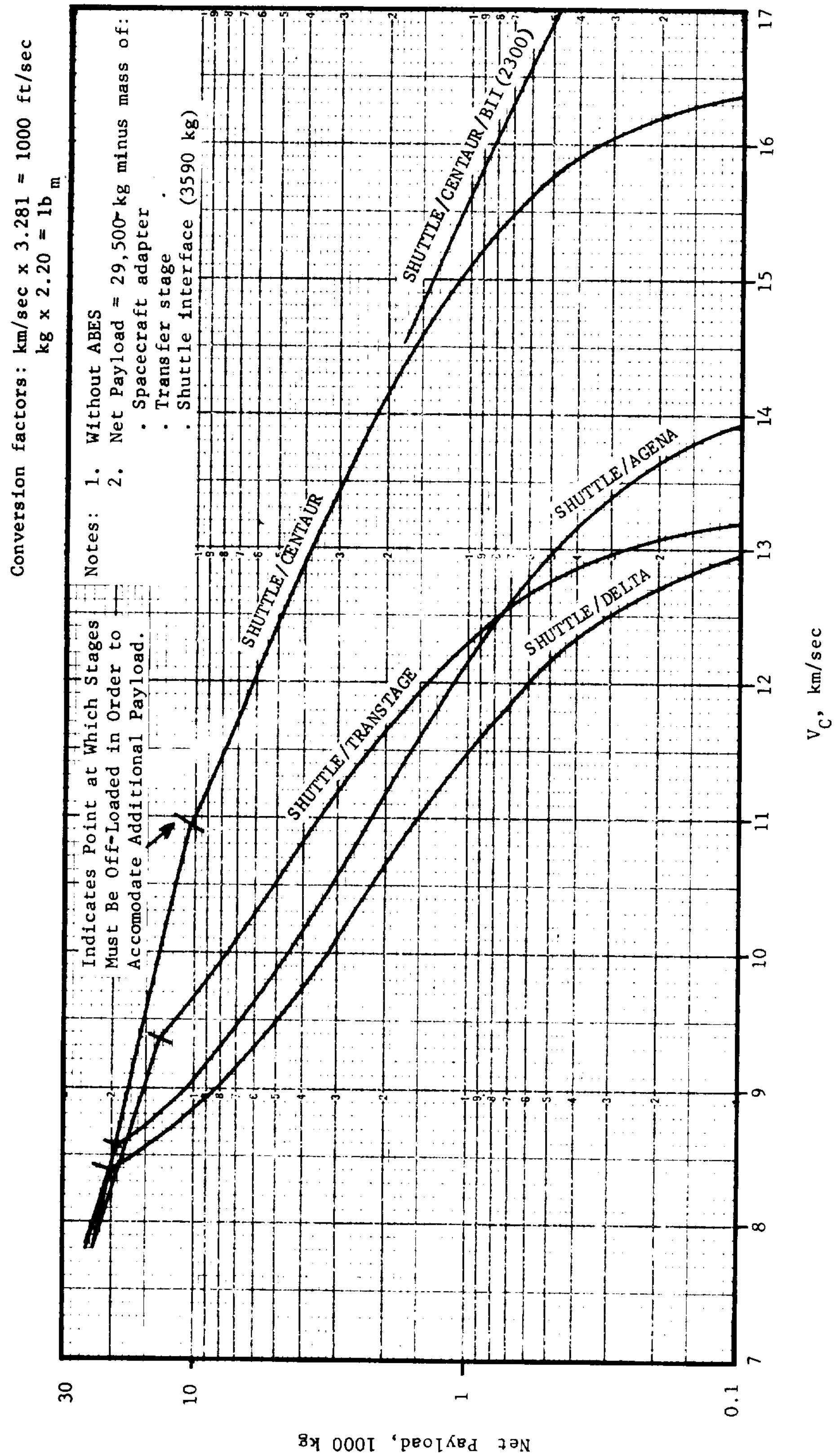


FIGURE 7-5. SPACE SHUTTLE PLUS TRANSFER STAGE PERFORMANCE WITHOUT AIR-BREATHING ENGINE SYSTEM FOR ORBITAL INCLINATIONS BETWEEN 28.5 AND 58 DEGREES

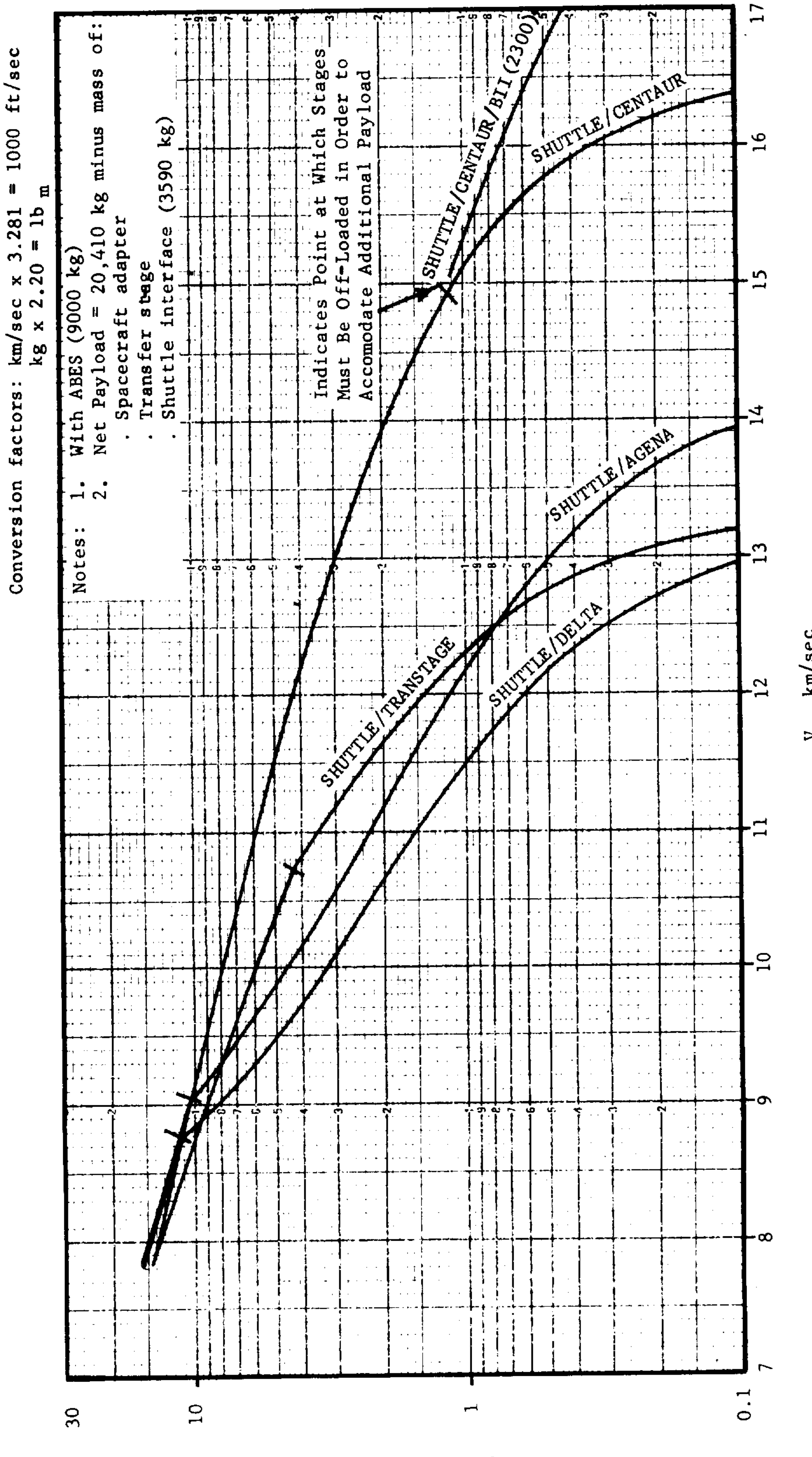


FIGURE 7-6. SPACE SHUTTLE PLUS TRANSFER STAGE PERFORMANCE WITH AIR-BREATHING ENGINE SYSTEM FOR ORBITAL INCLINATIONS BETWEEN 28.5 AND 58 DEGREES

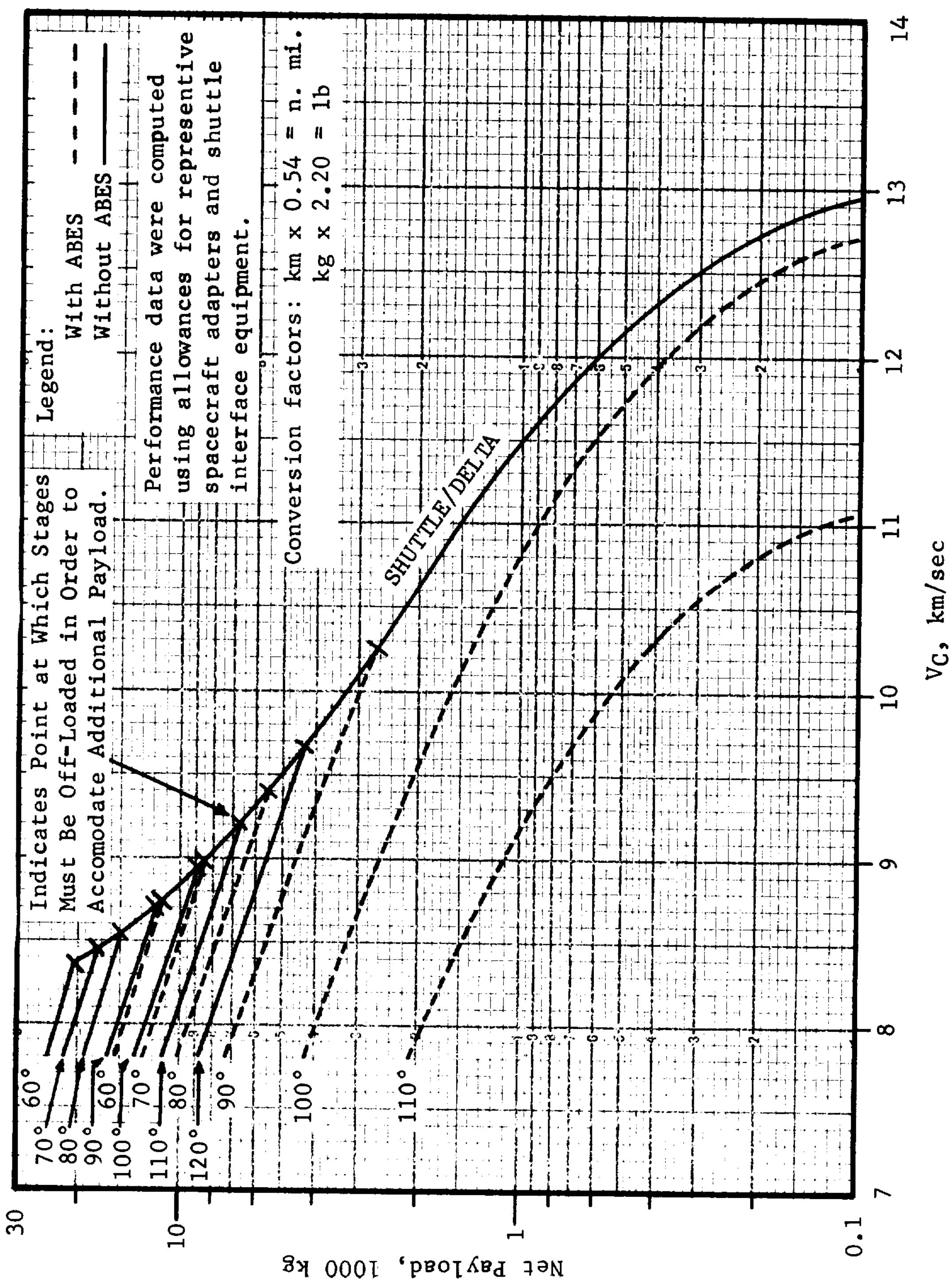


FIGURE 7-7. SHUTTLE/DELTA PERFORMANCE FOR NEAR POLAR ORBIT INCLINATIONS

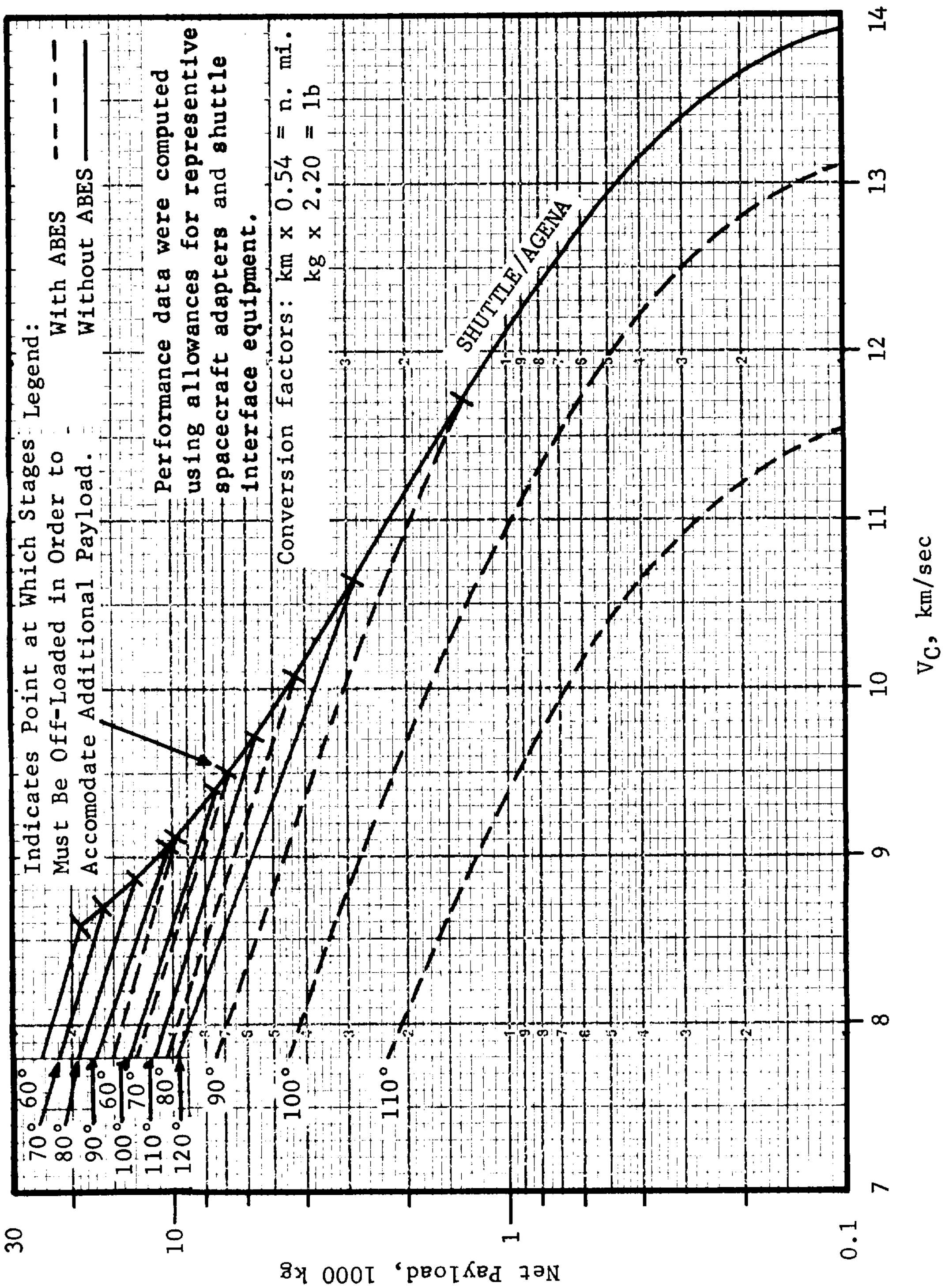


FIGURE 7-8. SHUTTLE/AGENA PERFORMANCE FOR NEAR POLAR ORBIT INCLINATIONS

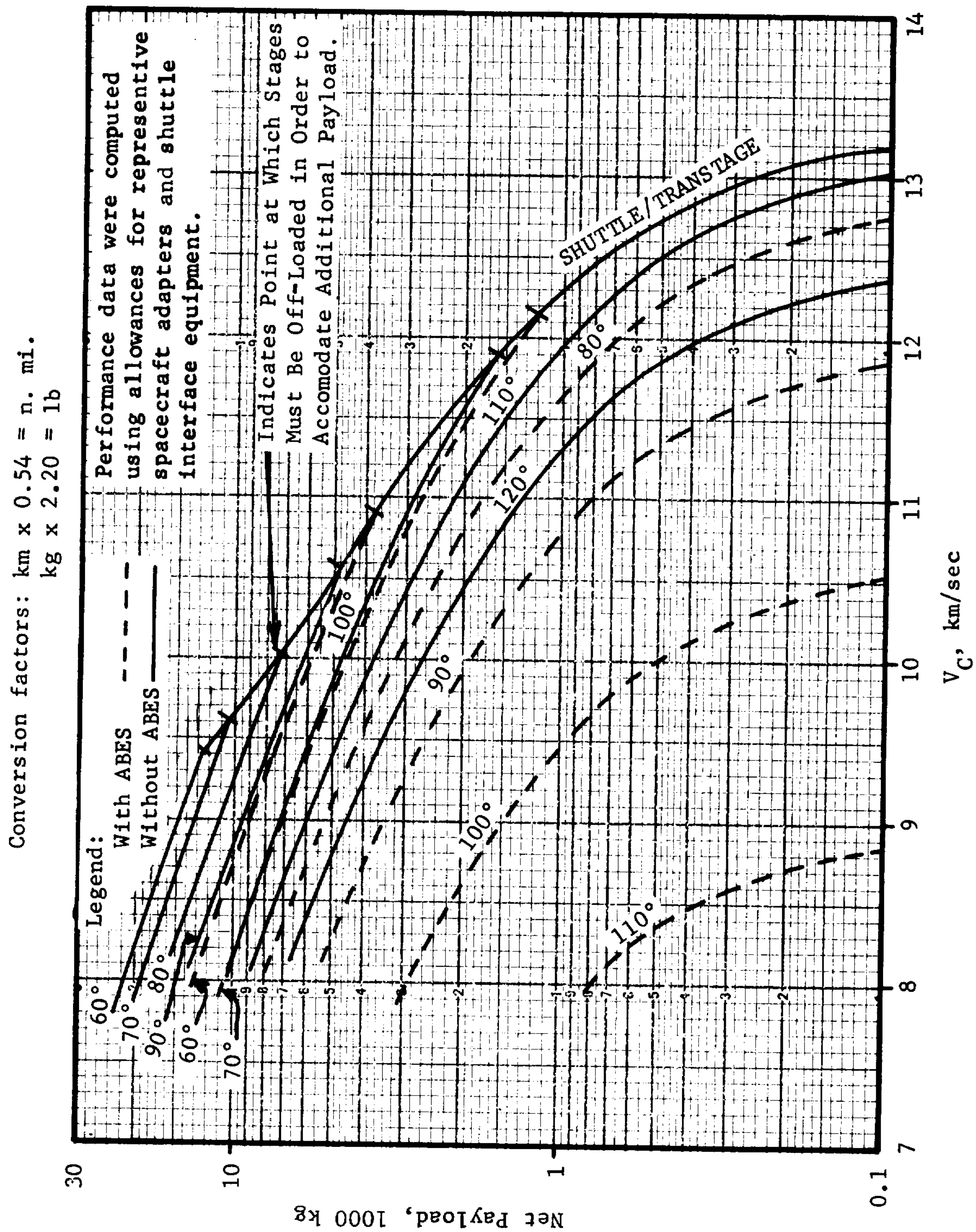


FIGURE 7-9. SHUTTLE/TRANSTAGE PERFORMANCE FOR NEAR POLAR ORBIT INCLINATIONS

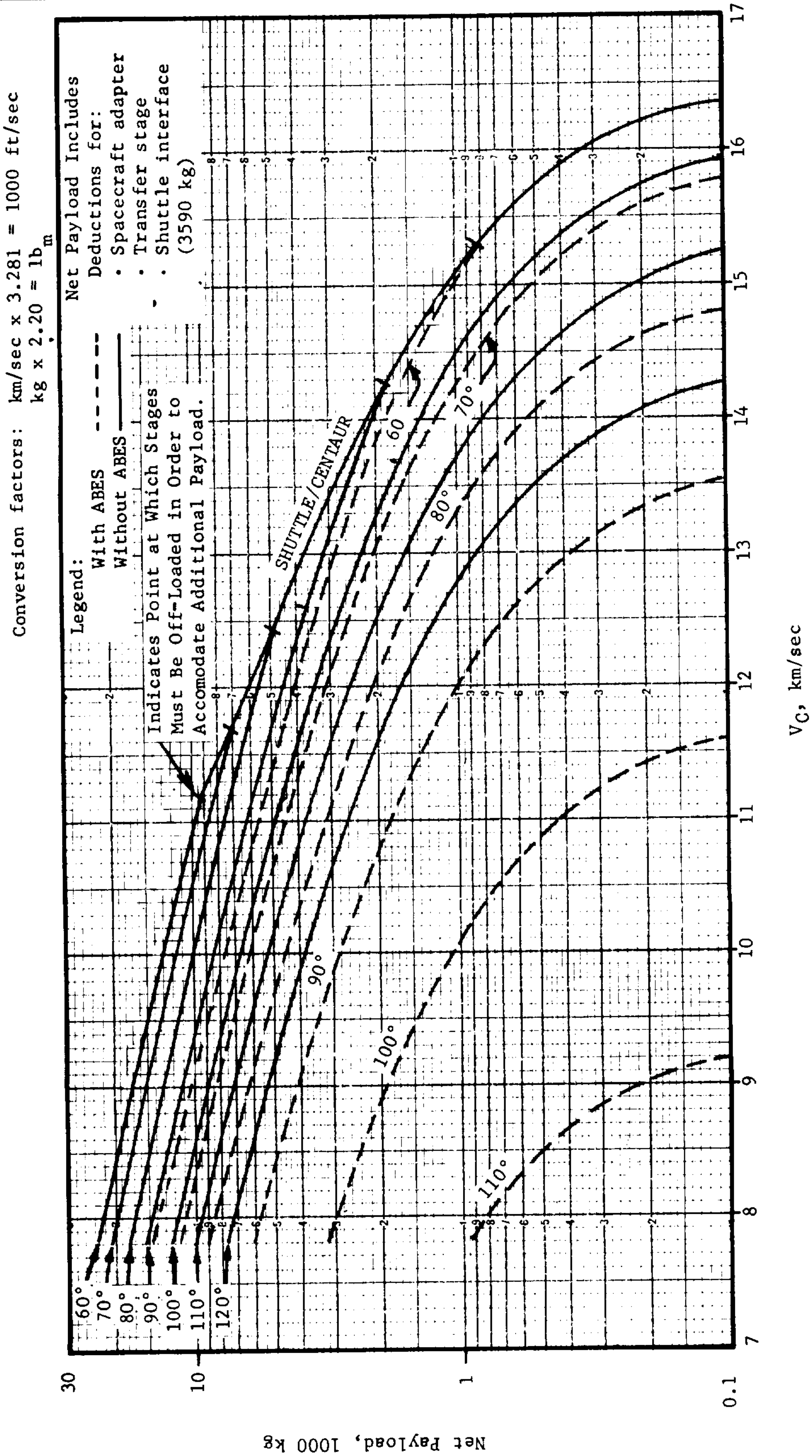


FIGURE 7-10. SHUTTLE/CENTAUR PERFORMANCE FOR NEAR POLAR ORBIT INCLINATIONS

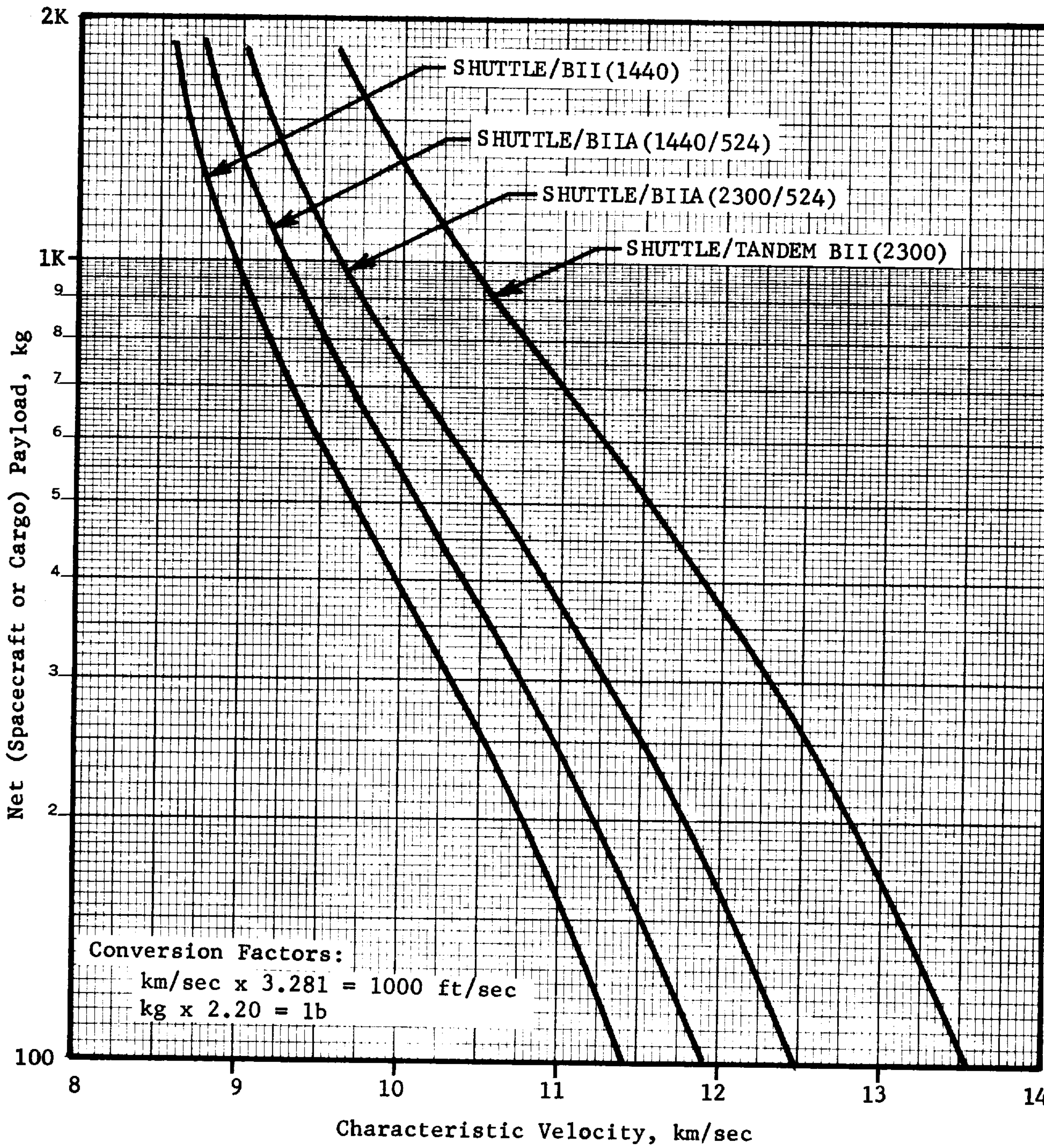


FIGURE 7-11. SHUTTLE/BURNER II TRANSFER STAGE PERFORMANCE FOR INCLINATIONS BETWEEN 28.5 AND 120 DEGREES

CHAPTER 8: VELOCITY-PACKAGE PERFORMANCE

TABLE OF CONTENTS

CHAPTER 8: VELOCITY-PACKAGE PERFORMANCE

800 GENERAL

In certain circumstances it may be possible to obtain higher characteristic velocities than provided by standard launch vehicles or the shuttle by including a small solid-propellant motor as an additional stage. Such motors with their associated control systems and structures are called velocity packages. Examples of velocity packages are the Burner II (BII) series and OV1 propulsion module (FW4 motor). The use of a velocity package is indicated, in general, only for payloads below the knee of the vehicle payload–characteristic velocity curve.

801 PROCEDURE

1. Table 8-1 gives characteristic data for several operational and proposed velocity packages. The FW4, TE364-3(1440), and TE364-4 (2300) velocity packages are spin stabilized and this must be considered in the design of the payload. The BIIA velocity packages consist of two stages mounted in tandem. The lower stage can be either a BII(1440), or a BII(2300) with the guidance and control systems removed. The upper stage is powered by a TE-M-442 motor with a propellant loading of 524 lb. The standard Burner II guidance and control systems are installed in the upper stage.

   Figure 8-1 presents total velocity increments as a function of payload for the velocity packages. Figure 8-2 shows the first and second-stage velocity increments provided by two-stage BIIA velocity packages. To use the information presented in this section for estimating the performance of vehicles with velocity packages, the following steps should be followed:

   a. Add the mass of the proposed payload to the total mass listed in Table 8-1 for the velocity package. If the payload mass exceeds the limit for which stage structure is designed, it will be necessary to account for the additonal mass of a strengthened structure. In this case, estimate the additional structural mass according to the procedures prescribed in the footnotes on Table 8-1 and include the additional mass in the summation of payload and velocity package mass.

b. Find the characteristic velocity of the proposed base vehicle from the curves in Chapter 5 for a payload equal to the total mass found in Step a.

c. Add to this characteristic velocity the velocity increment (ΔV) of the velocity package obtained from Figure 8-1 for the proposed payload mass. This gives total characteristic velocity available for the proposed payload using the velocity package.

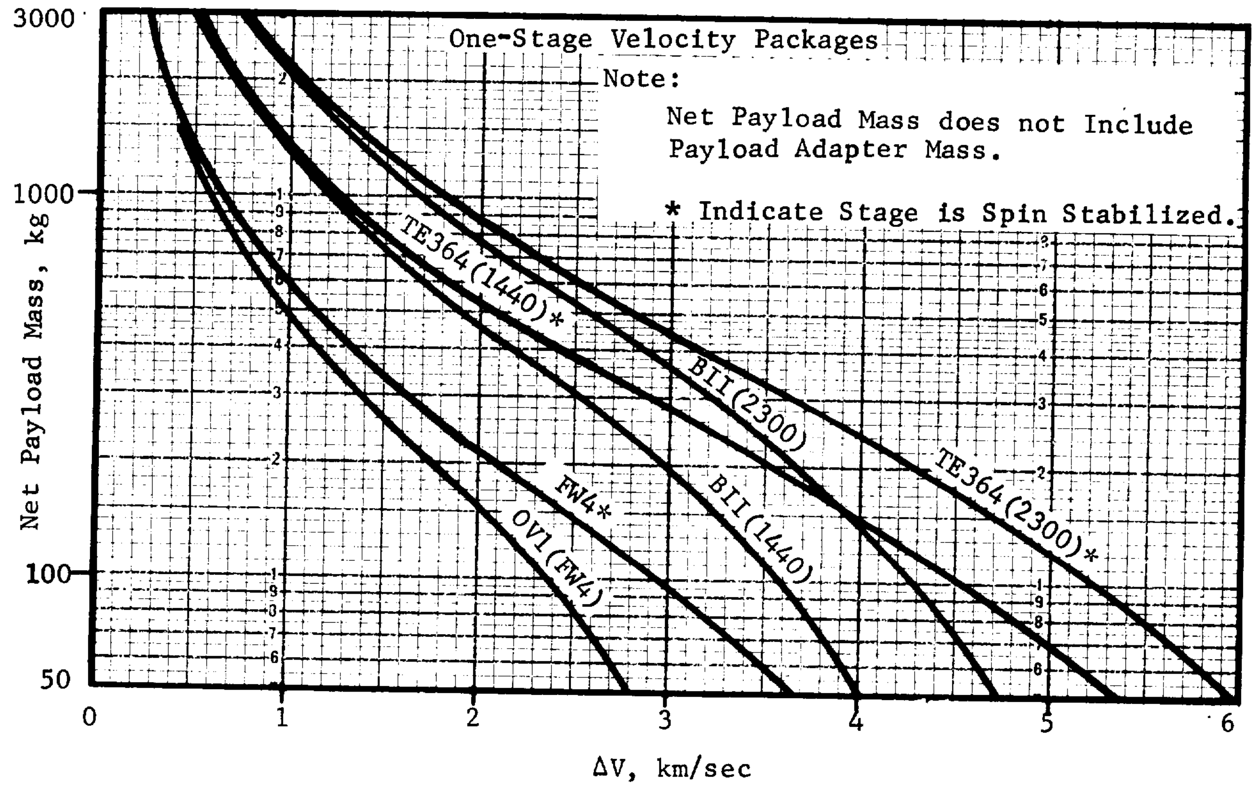
d. For Earth-orbital missions requiring two separate velocity increments, either a two-stage velocity package or a two-burn package would have to be used. The BIIA(1400/524) is the only operational two-stage velocity package, and no two-burn velocity packages are currently available. In considering the use of a two-stage velocity package Figure 8-2 should be consulted to determine whether the two velocity increments required by the mission can be provided by the velocity package. For an example of how Figure 8-2 is used, refer to subparagraph 704.3 e (page 7-14) and particularly subsection (2) (page 7-15). That sample problem demonstrates how Figure 8-2 is used in comparing individual stage performance with performance requirements determined from other parts of this book.

2. The user is cautioned that this procedure is an approximation. Variations in shroud masses and interstage adapters resulting from the use of velocity packages may cause the actual vehicle performance to differ from that estimated by this generalized procedure.

TABLE 8-1. VELOCITY-PACKAGE CHARACTERISTICS

Velocity Package	No. of Stages	Stabilization	Length/Diameter, m[b]	Stage Mass[a], kg[b]	Payload Adapter Mass[c], kg[b]	Nominal Maximum Payload Mass for Adapter, kg[b]	Status
TE364-3(1440)	1	Spin	1.4/1.0	715	9[c(1)]	250	Operational
BII(1440)	1	3-Axis	1.6/1.6	811	9[c(1)]	250	Operational
BIIA(1440/524)	2	3-Axis	2.3/1.6	1120	9[c(2)]	250	Operational
TE364-4(2300)	1	Spin	1.8/1.0	1092	9[c(3)]	500	Operational
BII(2300)	1	3-Axis	1.9/1.6	1233	9[c(3)]	500	Proposed
BIIA(2300/524)	2	3-Axis	2.9/1.6	1551	9[c(3)]	500	Proposed
OV1 (FW4)	1	3-Axis	2.1/0.6	390	-[c(4)]	100	Operational
FW4	1	Spin	1.6/0.5	316	9	--	Operational

(a) Stage masses do not include nominal payload adapter masses. The total velocity package mass is the sum of the stage mass and adapter mass.
(b) Conversion factors: m x 3.28 = ft, kg x 2.2 = lb.
(c) For greater than nominal payloads, the adapter mass must be increased as follows:
   (1) For payloads greater than 250 kg, add 10 percent of excess over 250 kg.
   (2) For payloads greater than 250 kg, add 12.5 percent of excess over 250 kg.
   (3) For payloads greater than 500 kg, add 10 percent of excess over 500 kg.
   (4) Adapter is integral with stage.



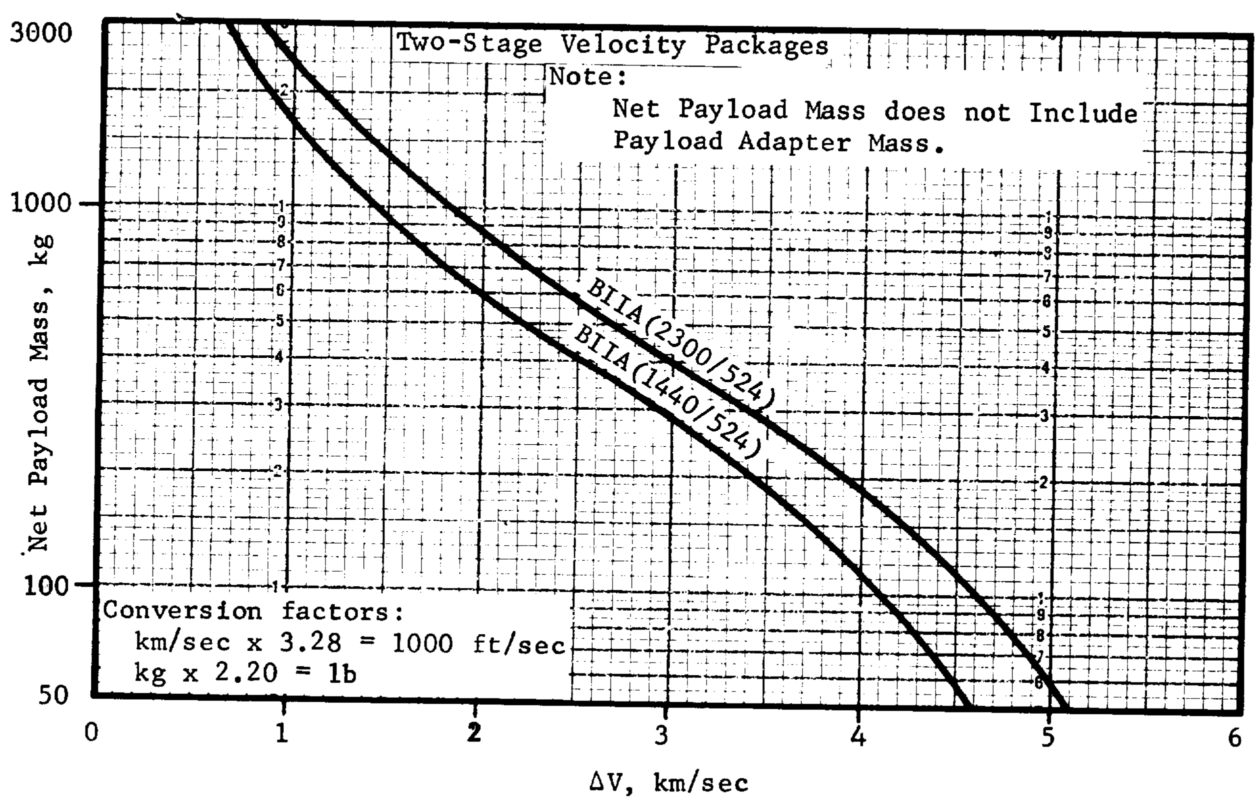


FIGURE 8-1. VELOCITY-PACKAGE PERFORMANCE

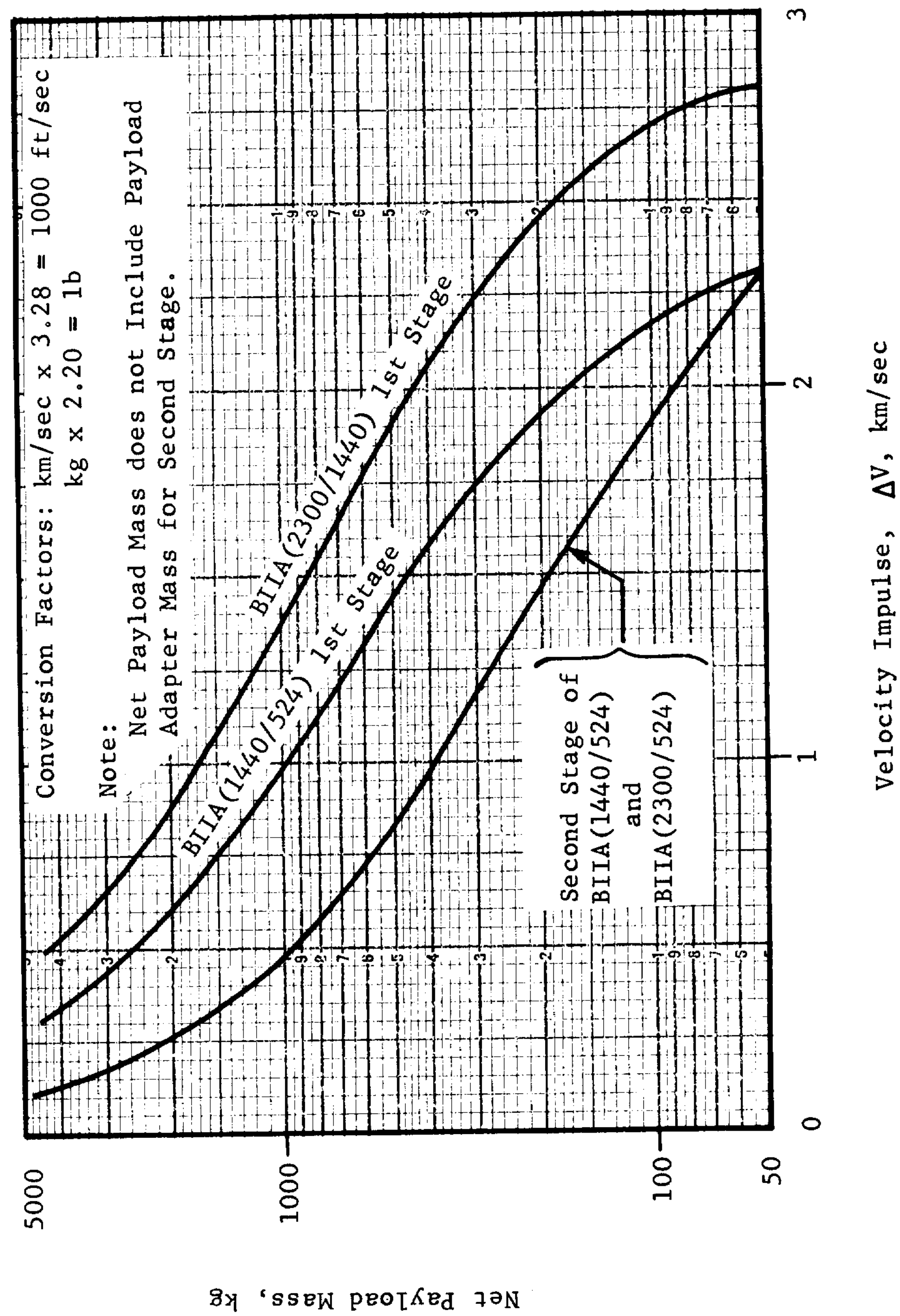


FIGURE 8-2. STAGE VELOCITY IMPULSES FOR TWO-STAGE SOLID PROPELLANT VELOCITY PACKAGES

CHAPTER 9: PERFORMANCE OF SOLAR-ELECTRIC PROPULSION SYSTEMS

TABLE OF CONTENTS

CHAPTER 9: PERFORMANCE OF SOLAR-ELECTRIC PROPULSION SYSTEMS

900 INTRODUCTION

This chapter contains performance data for selected launch vehicles or spacecraft containing a solar-electric propulsion (SEP) system. Solar-electric propulsion is discussed separately from other electric propulsion concepts because it is considered to be a nearer-term possibility. Data for other advanced propulsion concepts are shown in Chapter 10.

901 IDEALIZED-SYSTEM CAPABILITIES

1. Figures 9-1 through 9-10 show current estimates of possible planetary flyby- and orbiter-mission performance capabilities with spacecraft using solar-electric propulsion. Data are shown for a representative expendable launch vehicle (Titan IIIE/Centaur) and the space shuttle with a Centaur upper stage (Reference 32, Appendix B). No single general form has yet been developed to present performance data for low-thrust propulsion systems. Therefore, the performance of the low-thrust systems must be presented on a total vehicle and mission-by-mission basis.

2. The overall region of possible mission operations was determined by assuming a fully optimized propulsion system and by calculating the optimum constrained set of parameters for each mission and launch vehicle. This results in different values of power, exhaust velocity (I_{SP}), fuel mass, injected mass, injection energy, and thrust program at each point along the performance curve for each mission. This type of calculation indicates possible performance limits and the corresponding propulsion system parameters. The propulsion systems are based on present NASA programs oriented toward demonstrating the technology of small, solar-cell powered, ion engine propulsion systems. Assuming successful completion of this effort, operational launches in the late 1970's appear to be practical.

3. The performance curves are based on an assumption of circular coplanar planetary orbits and a propulsion system specific mass of 30 kg/kw at 1 a.u.; this mass includes the arrays, thrusters, and

power-conversion equipment. The net spacecraft mass (payload) is defined as the initial spacecraft weight minus the weight of the propulsion system (as defined above), propellant, tankage, and any structure that would not be required if the spacecraft propulsion system were not used. For orbiter missions, a highly eccentric elliptical orbit and a chemical retro-propulsion unit with an I_{SP} of 2940 m/sec and a propellant fraction of 0.9 for establishing the orbit have been assumed. For outer planet (Jupiter and beyond) and Mercury orbiters (Figures 9-5 to 9-8), it has been assumed that the solar-electric propulsion system will be jettisoned prior to the retro maneuver. For inner planet orbiters except those about Mercury (Figures 9-7 and 9-8), the solar-electric power supply is retained and goes into orbit with the spacecraft. However, the mass of the power supply is not included in the net spacecraft mass values shown on the figures.

4. Payloads for orbiter missions are highly dependent on the specification of the retro-propulsion system and on the capture orbit. For cases in which the basic launch vehicle outperforms the launch vehicle plus the solar-electric propulsion system, the basic launch-vehicle performance is shown by a broken line (e.g., Figure 9-3). The curves labeled with a "D" are for direct trajectories (transfer angle less than 360 degrees). Those labeled "I" are for indirect trajectories. The pairs of numbers at the ends of each curve indicate power available in kilowatts. One number indicates the value at 1 a.u. The other, in parentheses, indicates the power available at destination.

902 POTENTIAL OPERATIONAL SYSTEM CAPABILITIES

Although the fully optimized data of Figures 9-1 through 9-10 are useful for indicating the upper bound of expected performance, a more realistic analysis involves the selection of a limited number (ideally, one) of fixed designs and examining their performance for various missions. A number of concepts reflecting this approach are currently under investigation; these include multi-mission spacecraft with integrated solar-electric propulsion subsystems, attachable SEP propulsion modules, and fully independent SEP stages (References 33-34, Appendix B). The ultimate choice among these alternative approaches can depend on various factors including cost, performance, and programmatic and management considerations. Figures 9-11 and 9-12 illustrate the performance of the stage concept for a representative set of missions and two launch vehicles, the Titan IIIE/Centaur and the Space Shuttle. The power level for the stage concept has not been finally determined; the data shown on Figures 9-11 and 9-12 are for one representative example; that is, 21 kwe.

903 LAUNCH-OPPORTUNITY WIDTHS

Figure 9-13 shows an example of the variation of solar-electric payload with launch opportunity width for Jupiter flyby missions (Reference 35, Appendix B). The curves are for a specific flight time, mission, and launch vehicle and are based on specific SEP parameters. However, the general form of these curves is considered to be representative of a broad range of SEP missions. To an extent, the performance of a solar-electric system is dependent on desired opportunity width, but the dependence is much less severe than for ballistic trajectories.

Conversion Factor: Kilograms x 2.20 = lb_M

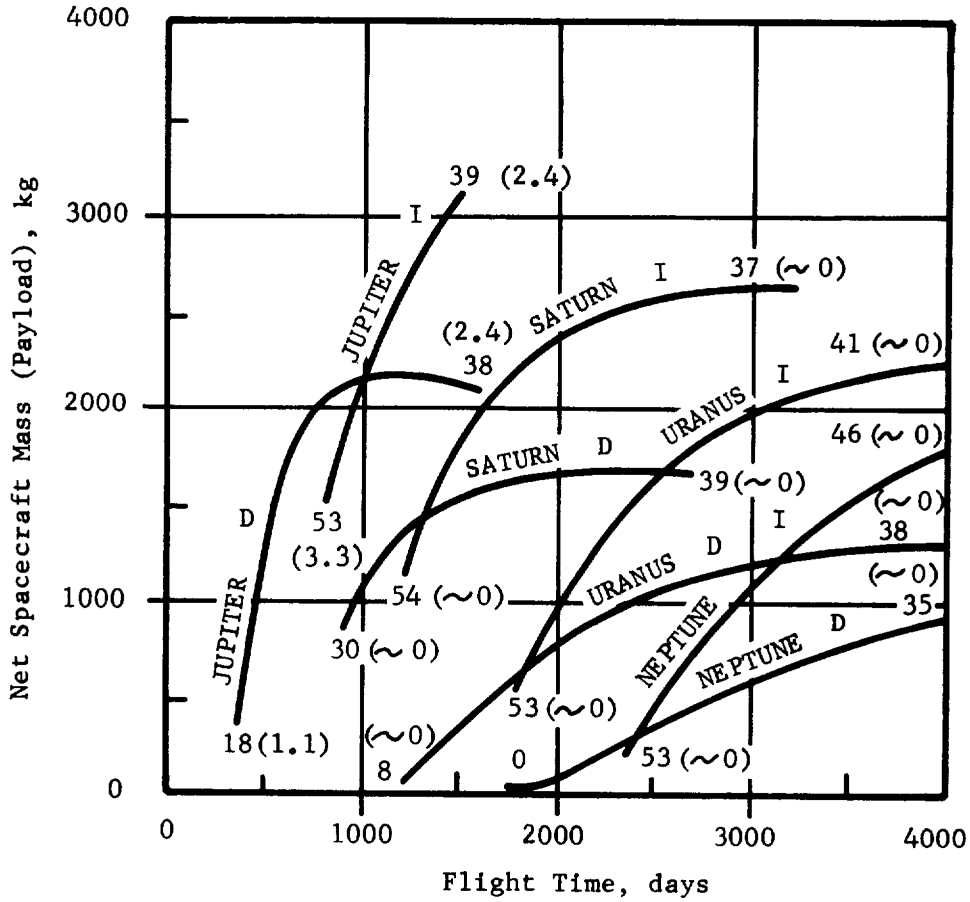


FIGURE 9-1. CAPABILITY OF A TITAN IIIE/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET FLYBYS

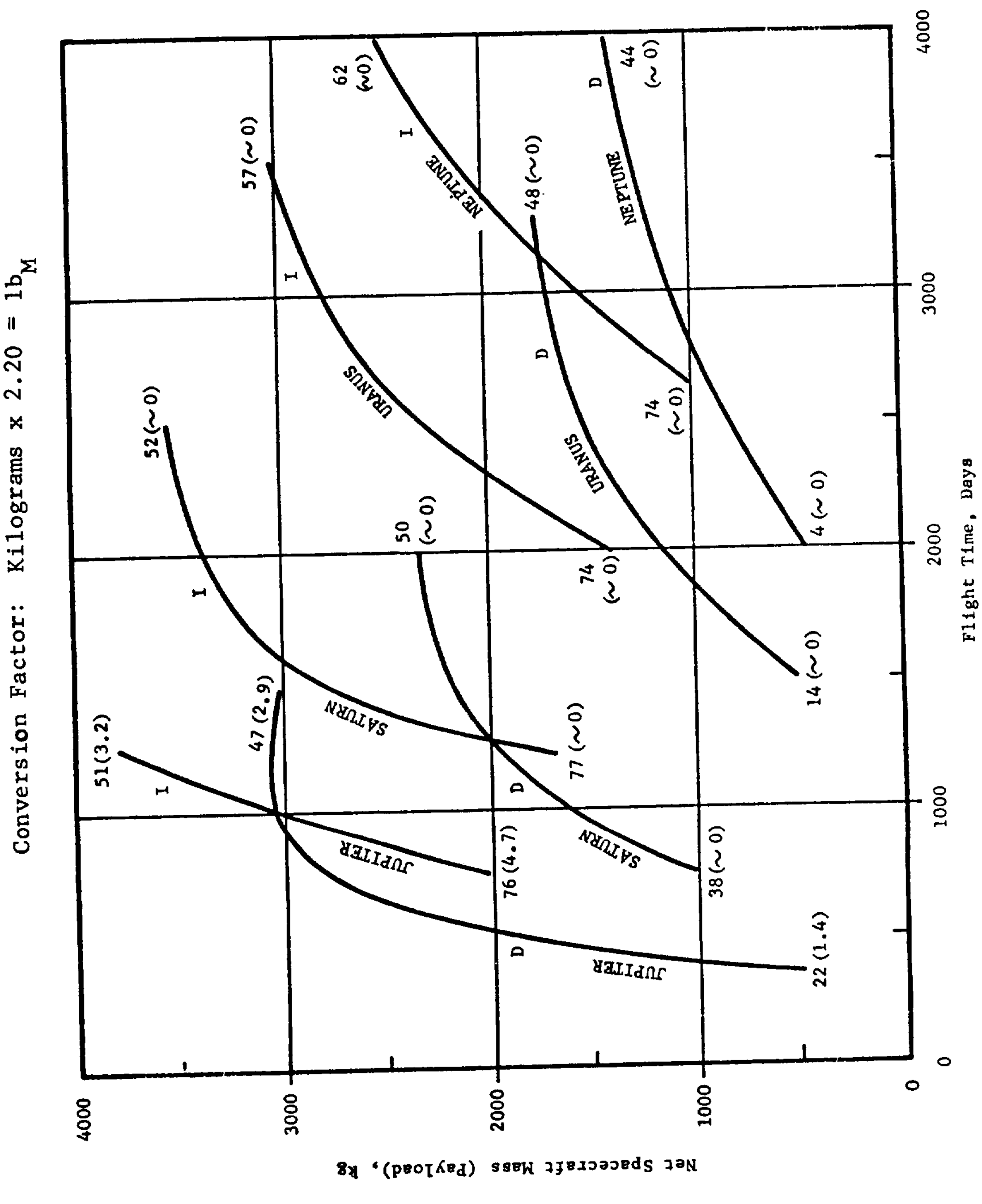


FIGURE 9-2. CAPABILITY OF A SPACE SHUTTLE/CENTAUR/ SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET FLYBYS

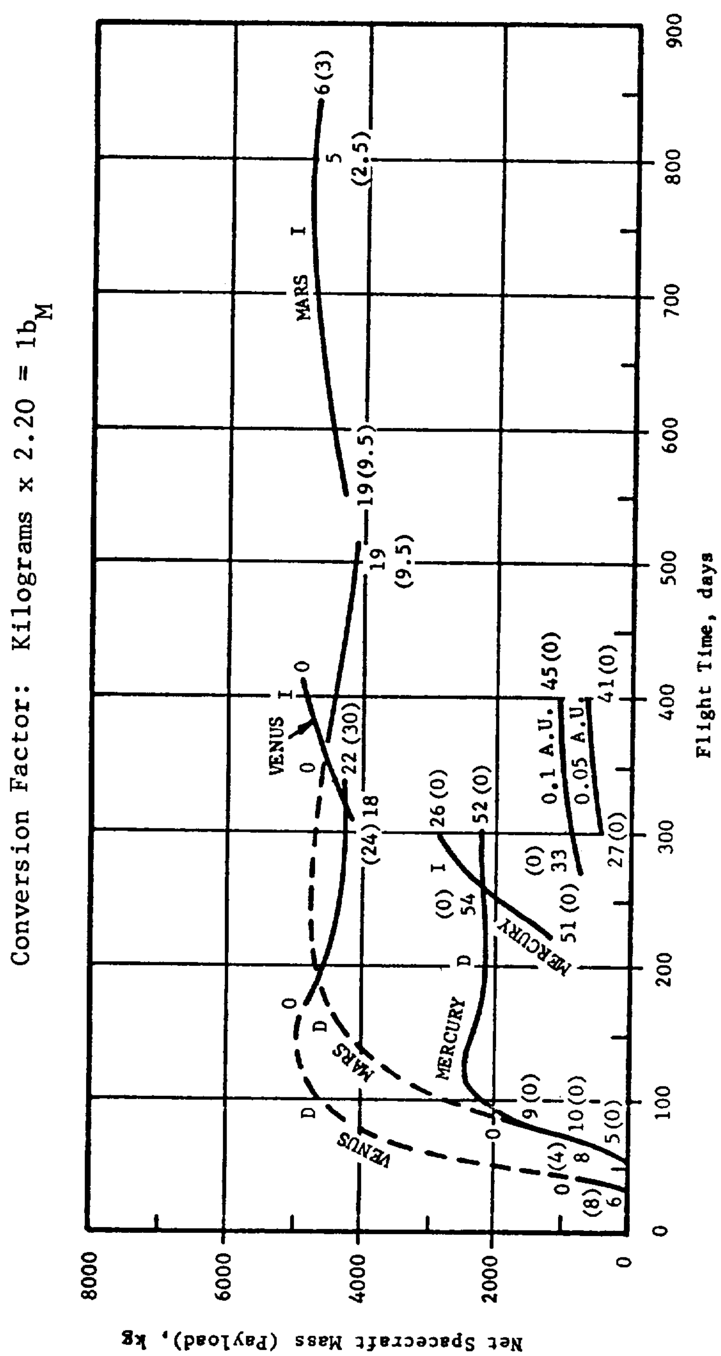


FIGURE 9-3. CAPABILITY OF A TITAN IIIE/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY FLYBYS AND CLOSE SOLAR PROBES

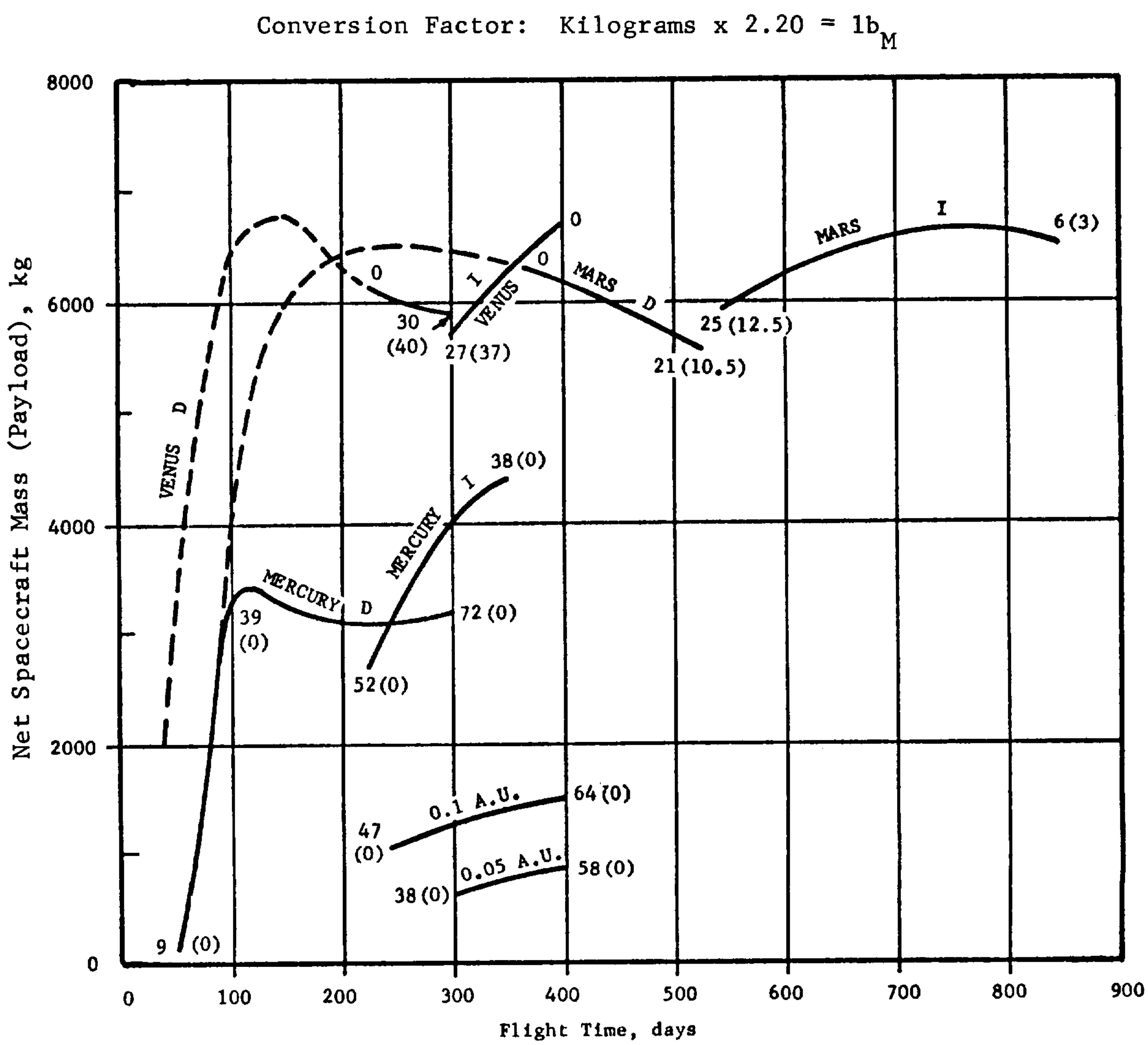


FIGURE 9-4. CAPABILITY OF A SPACE SHUTTLE/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY FLYBYS AND CLOSE SOLAR PROBES

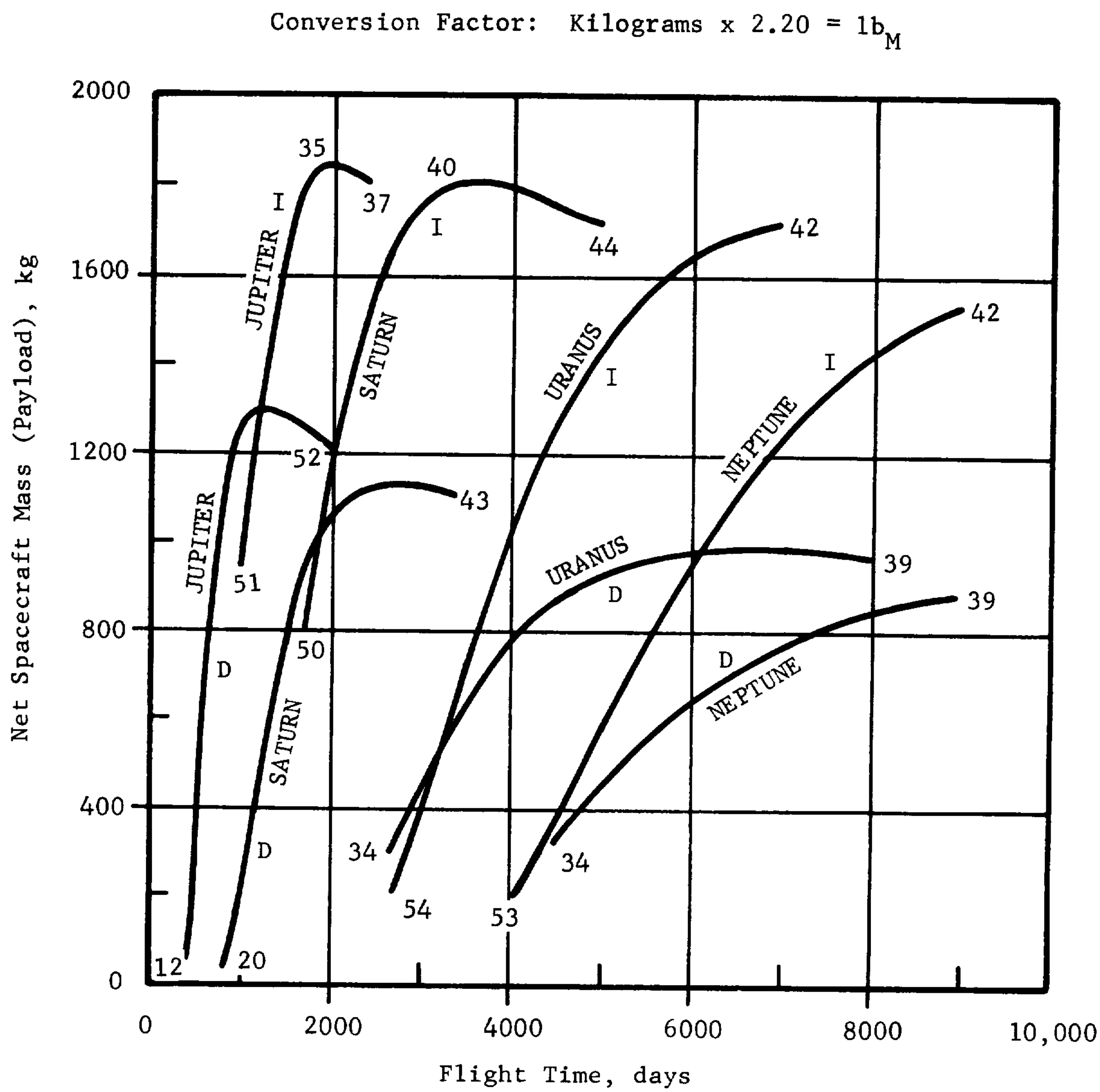


FIGURE 9-5. CAPABILITY OF A TITAN IIIE/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET ORBITERS

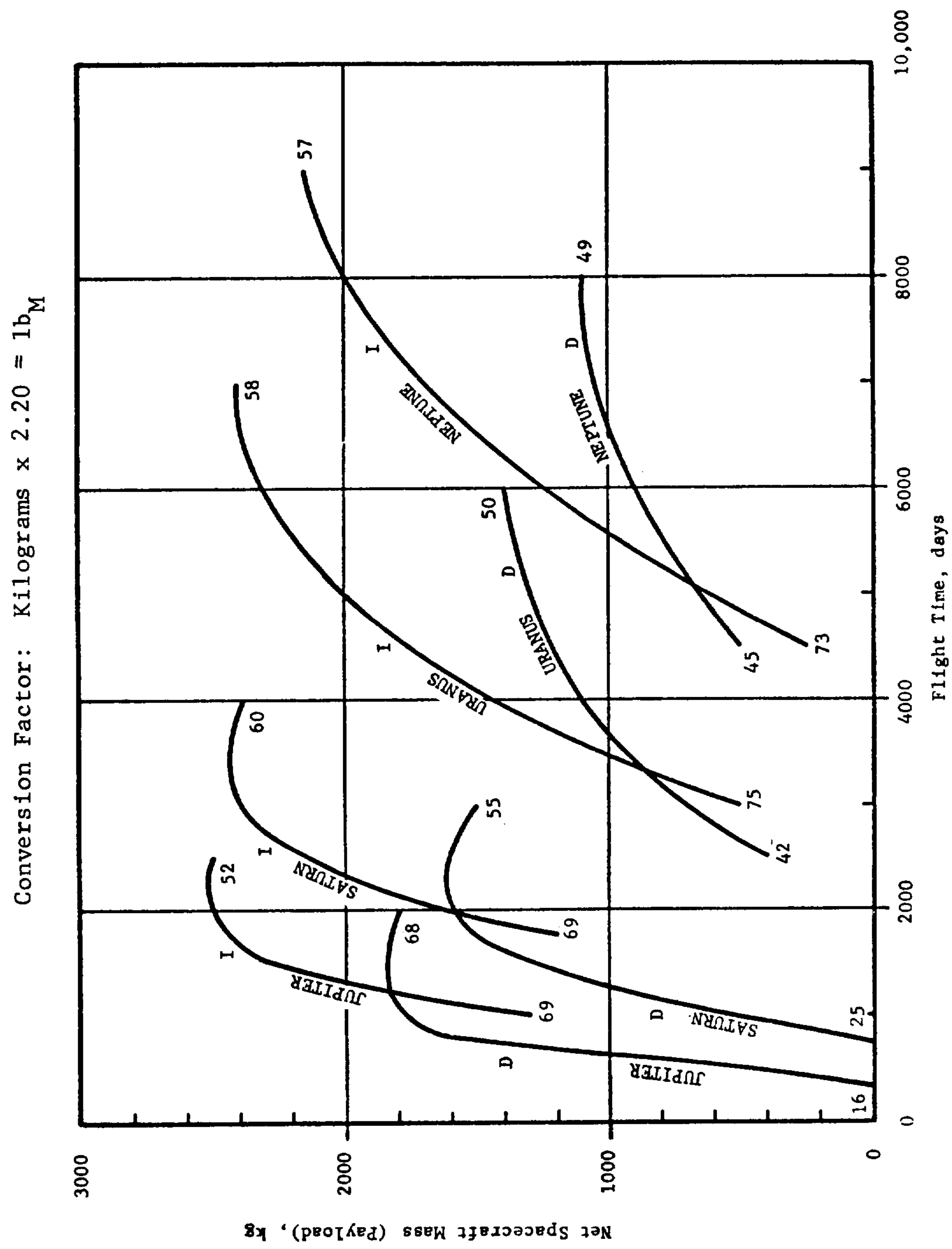


FIGURE 9-6. CAPABILITY OF A SPACE SHUTTLE/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET ORBITERS

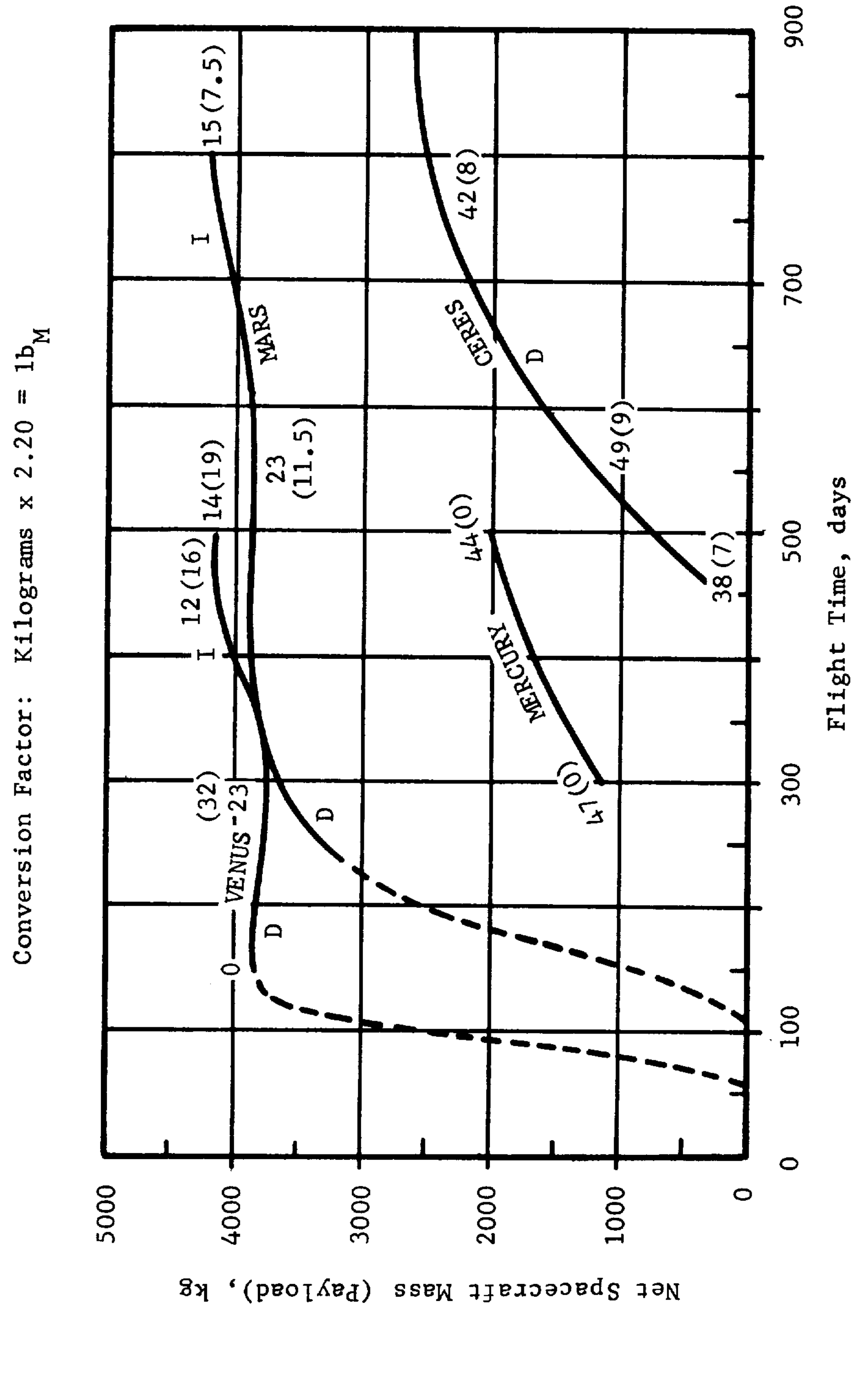


FIGURE 9-7. CAPABILITY OF A TITAN IIIE/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY ORBITER AND CERES RENDEZVOUS MISSIONS

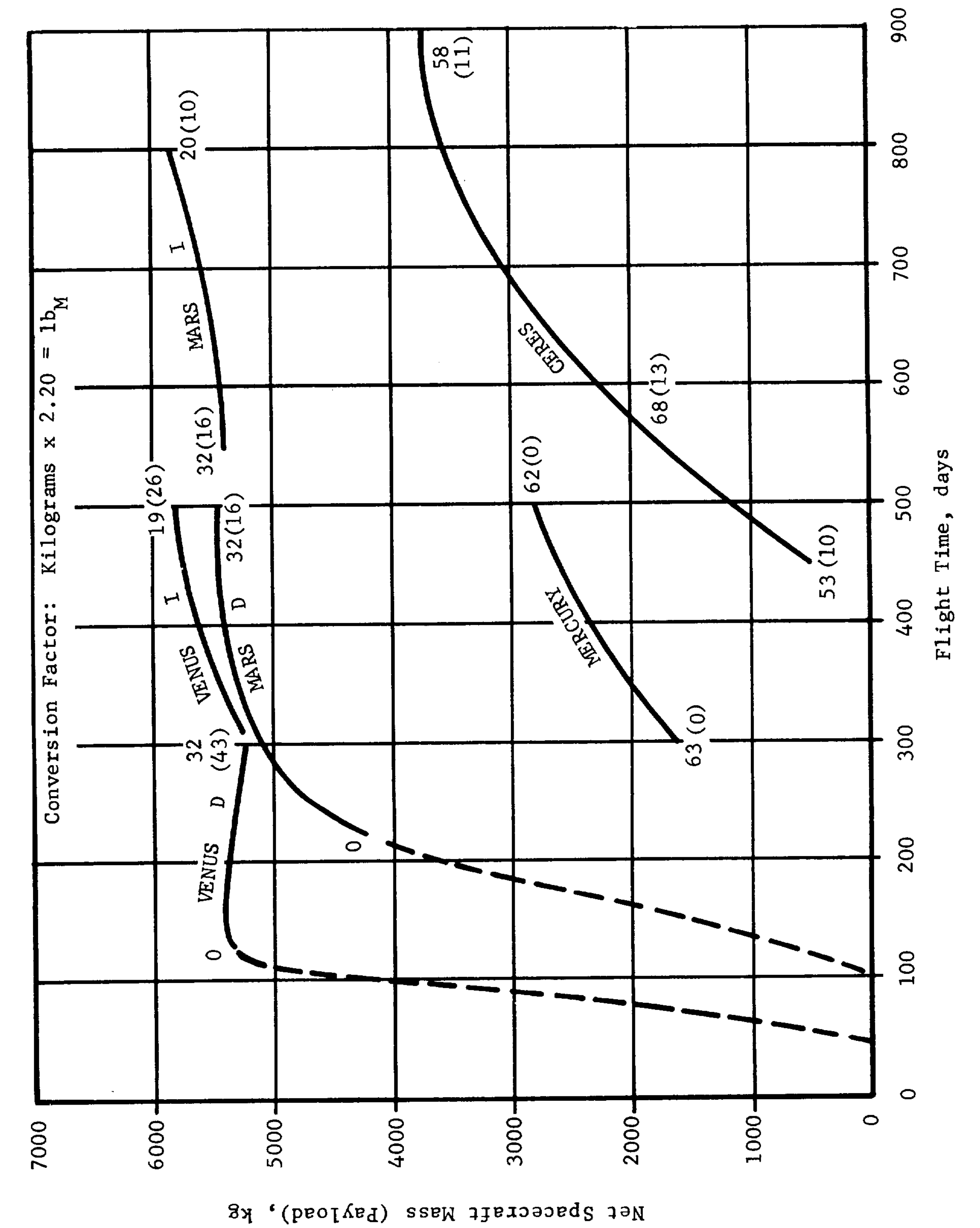


FIGURE 9-8. CAPABILITY OF A SPACE SHUTTLE/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY ORBITER AND CERES RENDEZVOUS MISSIONS

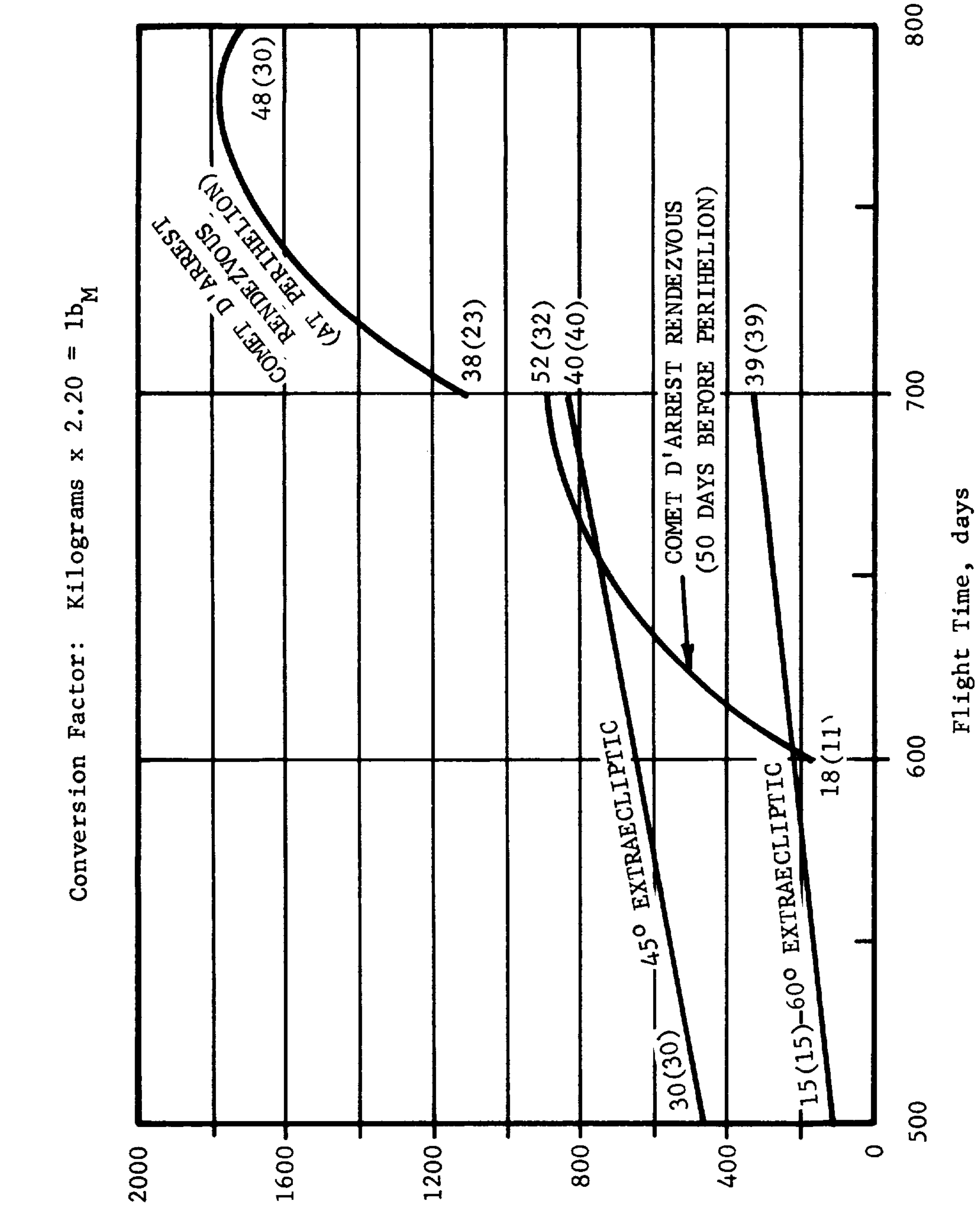


FIGURE 9-9. CAPABILITY OF A TITAN IIIE/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR COMETARY AND EXTRAECLIPTIC MISSIONS

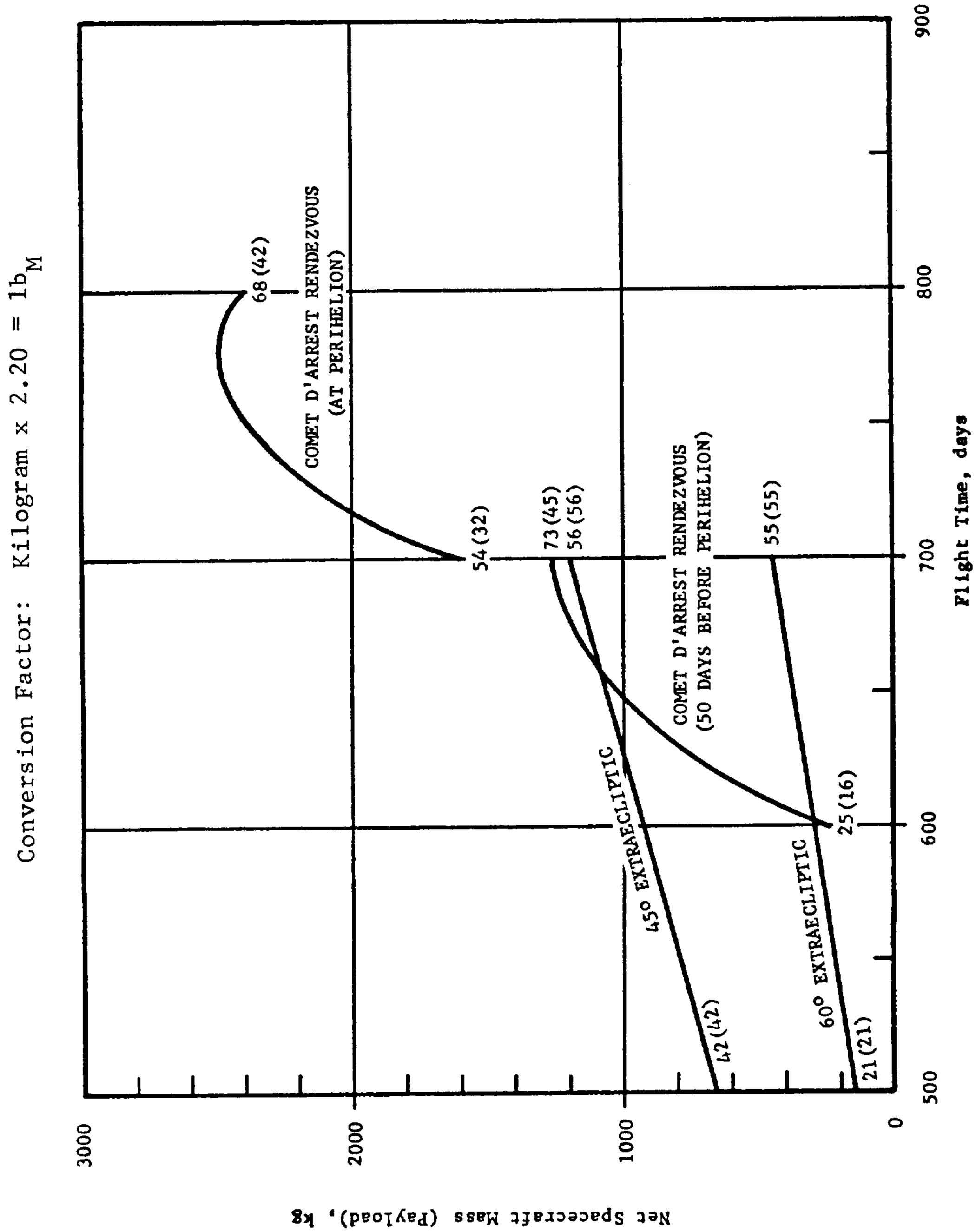


FIGURE 9-10. CAPABILITY OF A SPACE SHUTTLE/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR COMETARY AND EXTRAECLIPTIC MISSIONS

Conversion Factor: Kilograms x 2.20 = lb_M

1. MERCURY ORBITER (2 x 25)
2. VENUS ORBITER (2 x 18)
3. JUPITER ORBITER (4 x 82)
4. SATURN ORBITER (4 x 44)
5. URANUS ORBITER/PROBE (1.1 x 64, 240 kg PROBE)
6. ENCKE RENDEZVOUS (50 DAYS BEFORE PERIHELION)
7. VESTA RENDEZVOUS
8. PLUTO FLYBY

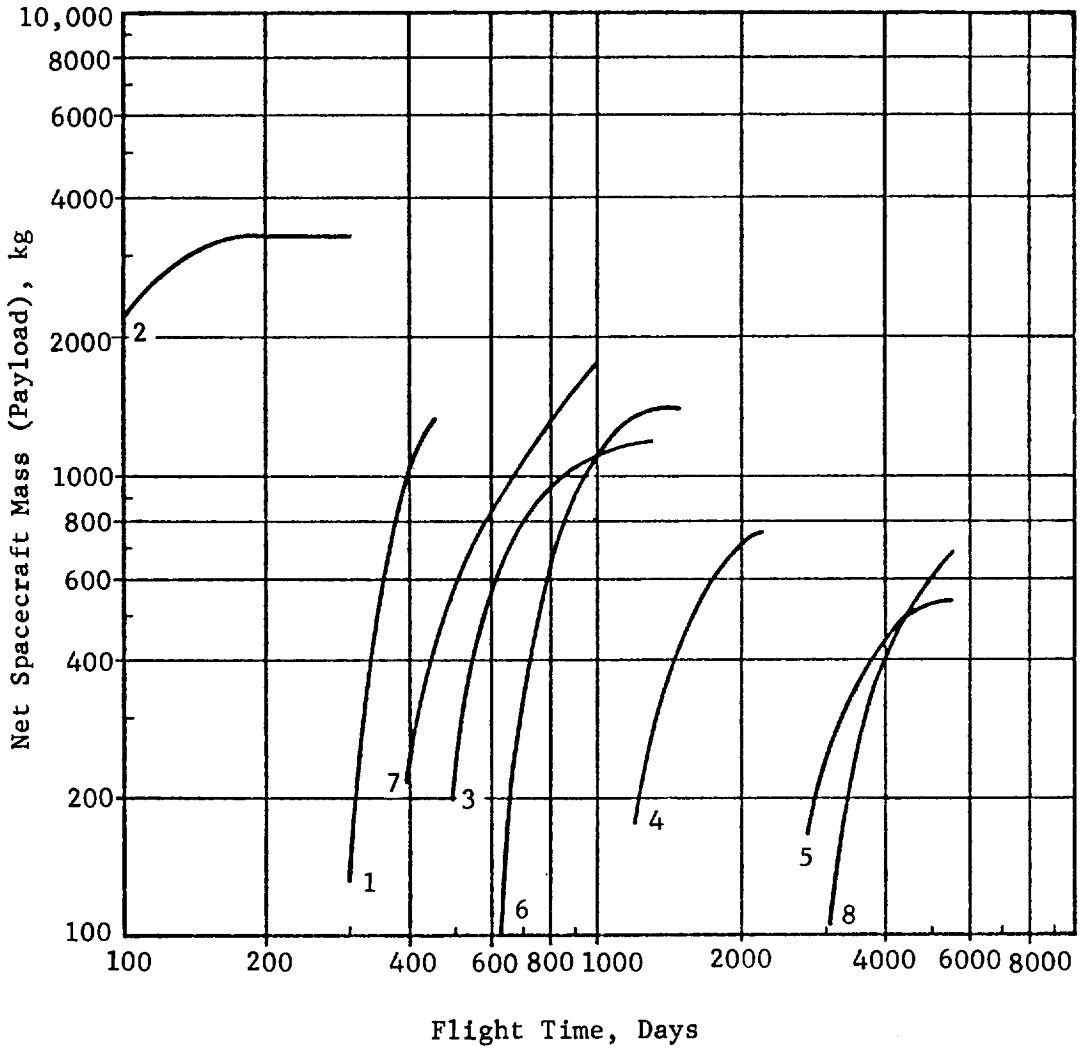


FIGURE 9-11. PERFORMANCE OF TITAN IIIE/CENTAUR/SOLAR-ELECTRIC STAGE (21 KWE) LAUNCH VEHICLE

Conversion Factor: Kilograms x 2.20 = lb_M

1. MERCURY ORBITER (2 x 25)
2. VENUS ORBITER (2 x 18)
3. JUPITER ORBITER (4 x 82)
4. SATURN ORBITER (4 x 44)
5. URANUS ORBITER/PROBE (1.1 x 64, 240 kg PROBE)
6. ENCKE RENDEZVOUS (50 DAYS BEFORE PERIHELION)
7. VESTA RENDEZVOUS
8. PLUTO FLYBY

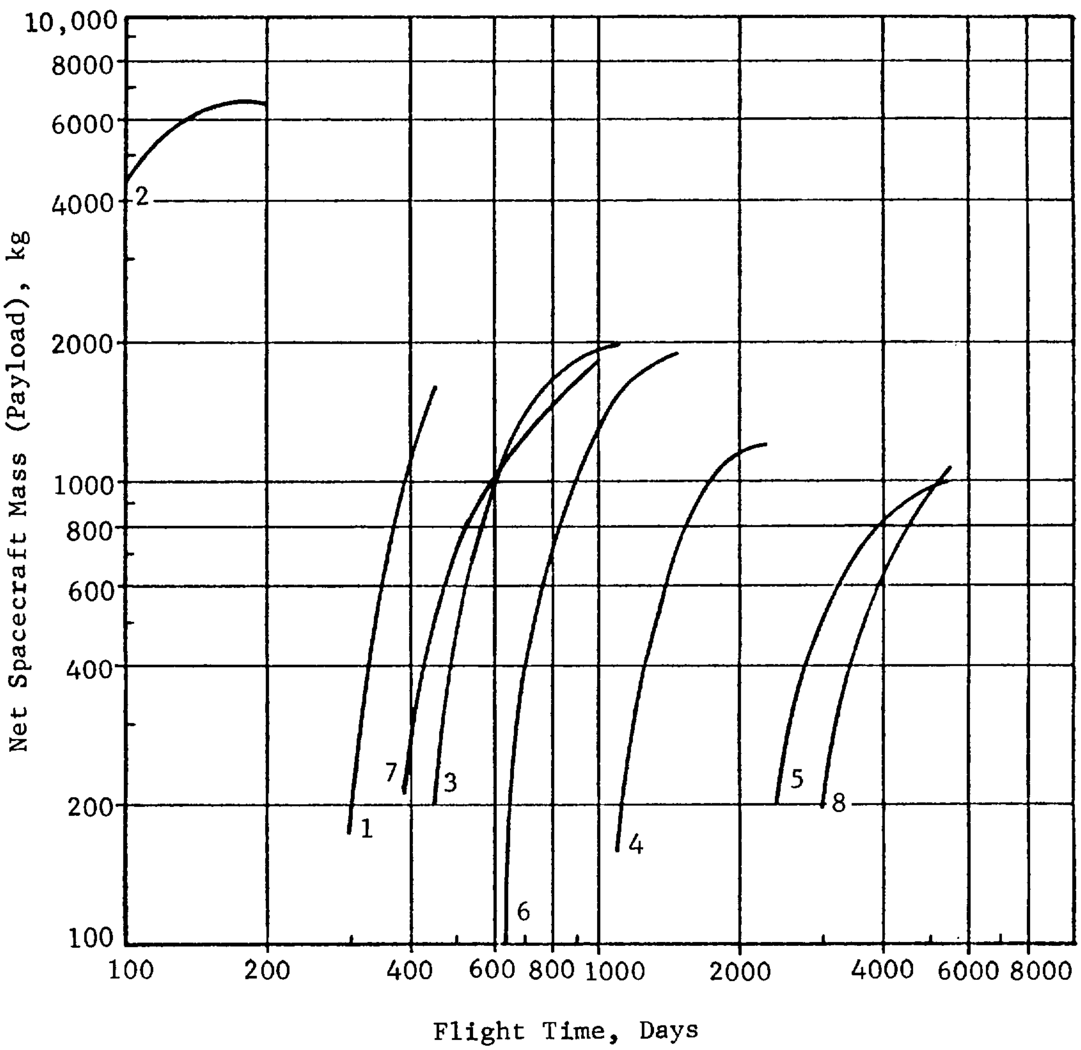


FIGURE 9-12. PERFORMANCE OF SHUTTLE/CENTAUR/SOLAR-ELECTRIC STAGE (21 KWE) LAUNCH VEHICLE

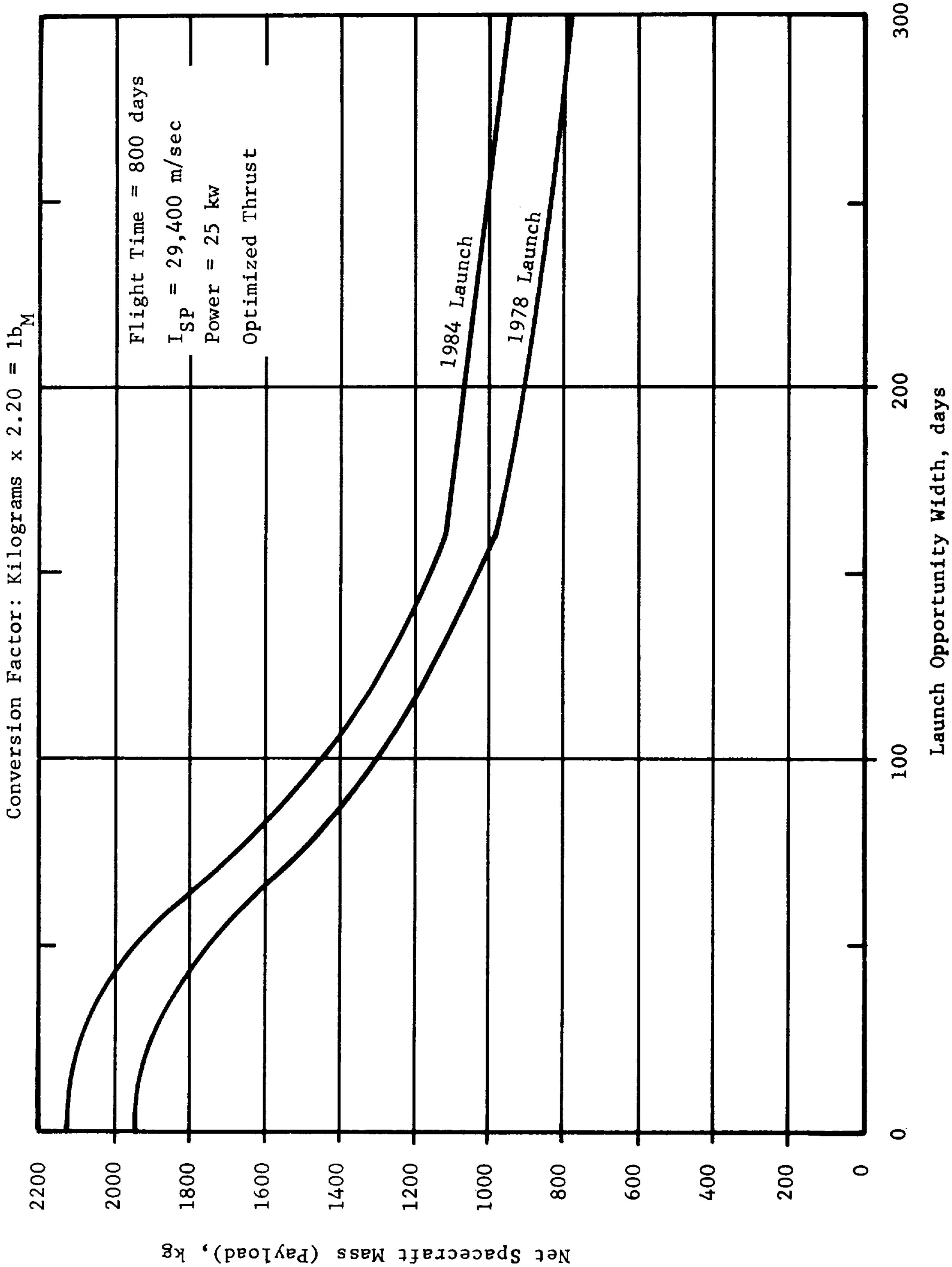


FIGURE 9-13. OPPORTUNITY WIDTH FOR JUPITER FLYBY MISSIONS USING A TITAN IIIE/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE

CHAPTER 10: PERFORMANCE OF NUCLEAR-THERMAL AND NUCLEAR-ELECTRIC PROPULSION SYSTEMS

TABLE OF CONTENTS

CHAPTER 10: PERFORMANCE OF NUCLEAR-THERMAL AND NUCLEAR-ELECTRIC PROPULSION SYSTEMS

1001 INTRODUCTION

1. The figures in this chapter present performance data for selected launch vehicles with upper stages or spacecraft equipped with nuclear-thermal or nuclear-electric propulsion systems. The systems shown were selected on the basis of their potential for improving mission capability.

1002 NUCLEAR-THERMAL PROPULSION SYSTEMS

1. Figures 10-1 and 10-2 show performance estimates for the small nuclear stage with the Alpha and Gamma nuclear-thermal engines, respectively, launched using the shuttle (References 36-37, Appendix B). The specific impulse of the Alpha engine is estimated at 8500 m/sec, while that of the Gamma engine is projected at 9560 m/sec. These curves are based on using all allowable propellant in a single engine operation, with gravity losses and near-optimal steering benefits included. Performance curves are included for proposed vehicles assembled from components carried to orbit by two or three shuttle launches. Some curves also include the effects of using a kick stage with space storable propellant. If a retro-system is required at the destination, it must be considered part of the payload in these two figures.

2. The nuclear-thermal stage is assumed to start from a 435-km circular orbit, having been carried there by the space shuttle. The characteristic velocity is referenced to this altitude. Propellant is off-loaded for heavier payloads to remain within shuttle payload limits. If the shuttle is not used, the currently defined nuclear-thermal stage will require an expendable launch vehicle with capabilities great than any existing Titan III vehicles.

3. A proposed option would allow the nuclear-thermal stage to be restarted at the destination, thus permitting it to perform retro- or capture-maneuvers. Another proposed option allows the nuclear-thermal stage to provide 25 kwe of electrical power for payload or nuclear stage use. These two options are independent and may be combined.

4. The nuclear stage with the Alpha version of the nuclear-thermal engine might be made available by the end of the 1970's.

1003 NUCLEAR-ELECTRIC PROPULSION SYSTEMS

1. Figures 10-3 through 10-6 show the potential performance of nuclear-electric propulsion systems on the Titan IIIE/Centaur, the Titan III7/Centaur, the space shuttle, and the Shuttle/Centaur for several representative missions. The shuttle data used for these calculations are based on a slightly different version of the shuttle than that which is discussed in Chapter 7. However, the data are considered representative of expected shuttle performance. An assumed propulsion system specific mass of 30 kg/kwe for the 100-kwe system and 25 kg/kwe for the 250-kwe system was used in generating the performance estimates (Reference 38, Appendix B). The propulsion-system mass includes the power source, thrusters, and power-conversion equipment and is based on current technology estimates. The specific mass is primarily a function of the power level. Payload is defined as the initial spacecraft mass minus the mass of the propulsion system, propellant, tankage, and any structure that would not be required if the spacecraft propulsion system were not used. Spiral capture at the target planet has been assumed. The data shown are optimized; i.e., for each objective and flight time, a specific set of spacecraft parameters, specific impulse, booster injection velocity, etc., and a thrusting history have been chosen to yield the maximum delivered payload.

2. Some performance data on nuclear-electric propulsion systems for Mars and Eros sample return missions are available (Reference 38, Appendix B). This information indicates the feasiability of missions with durations of 600 to 1000 days and return payloads of approximately 450 kg. Systems discussed in Reference 38 include those requiring 1 or 2 shuttle launches, 100 or 250 kwe NEP systems (with or without chemical third stages, and NEP or NEP/chemical Earth return capture).

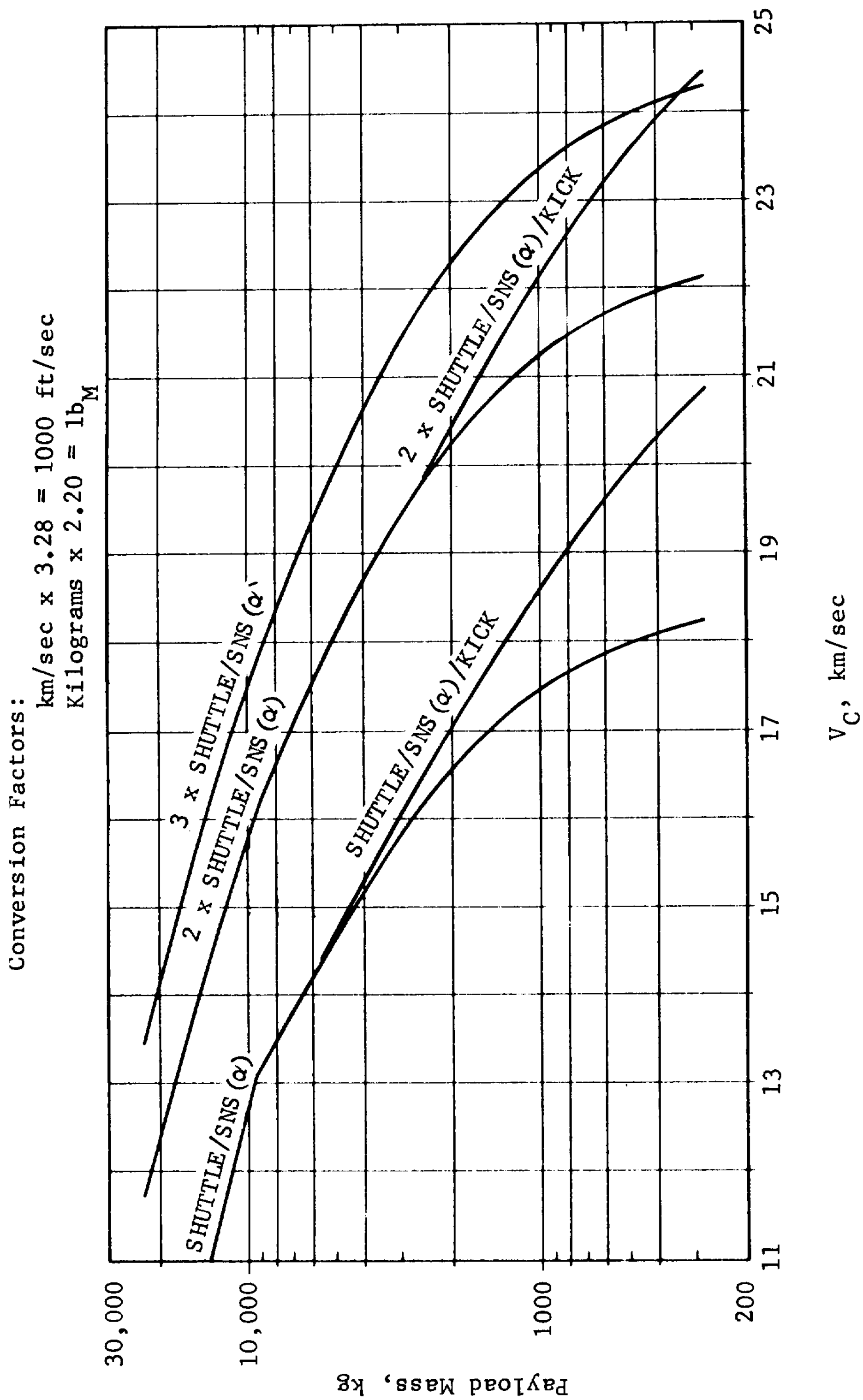


FIGURE 10-1. PERFORMANCE OF THE SMALL NUCLEAR STAGE (SNS)-ALPHA ENGINE

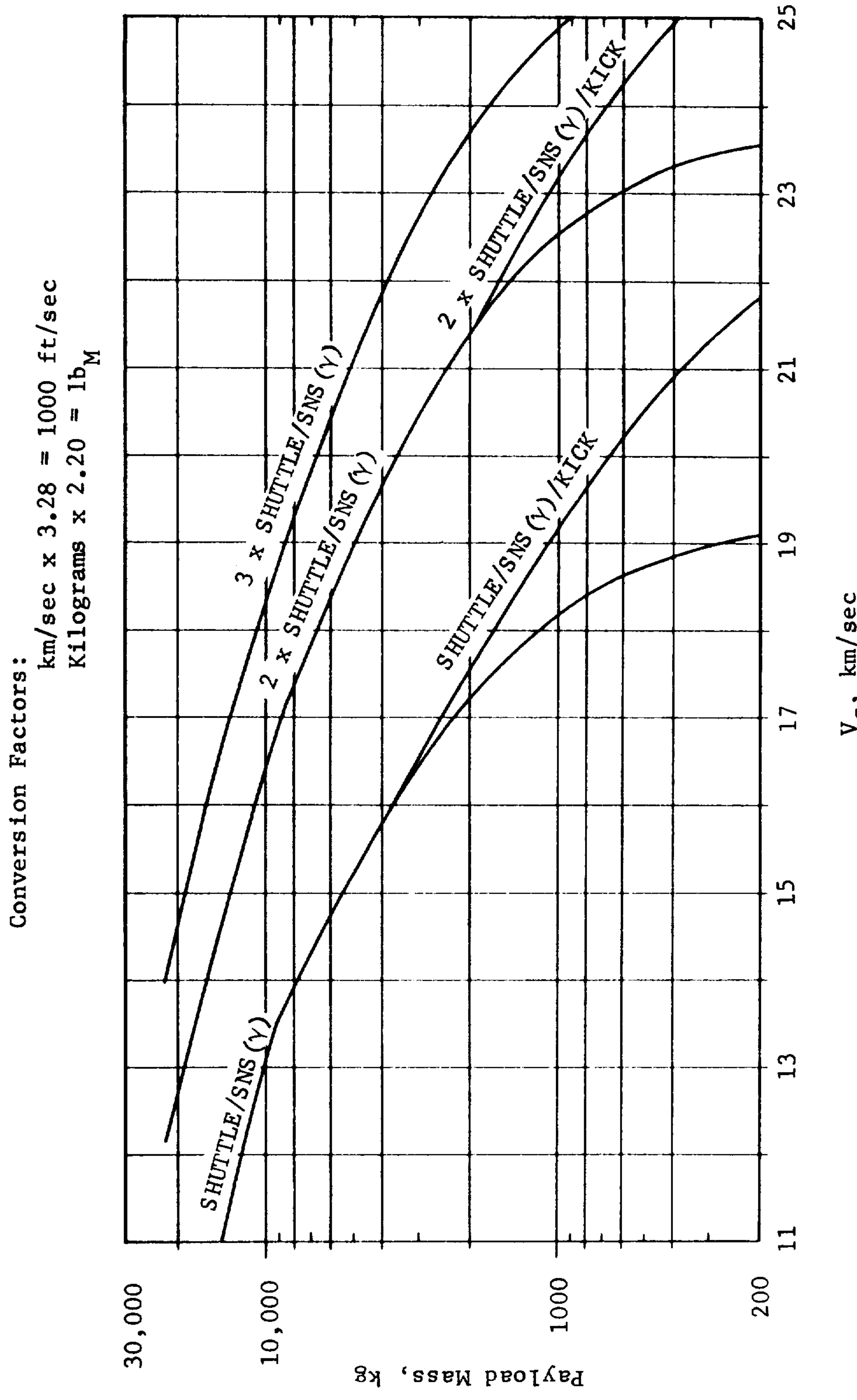


FIGURE 10-2. PERFORMANCE OF THE SMALL NUCLEAR STAGE (SNS)-GAMMA ENGINE

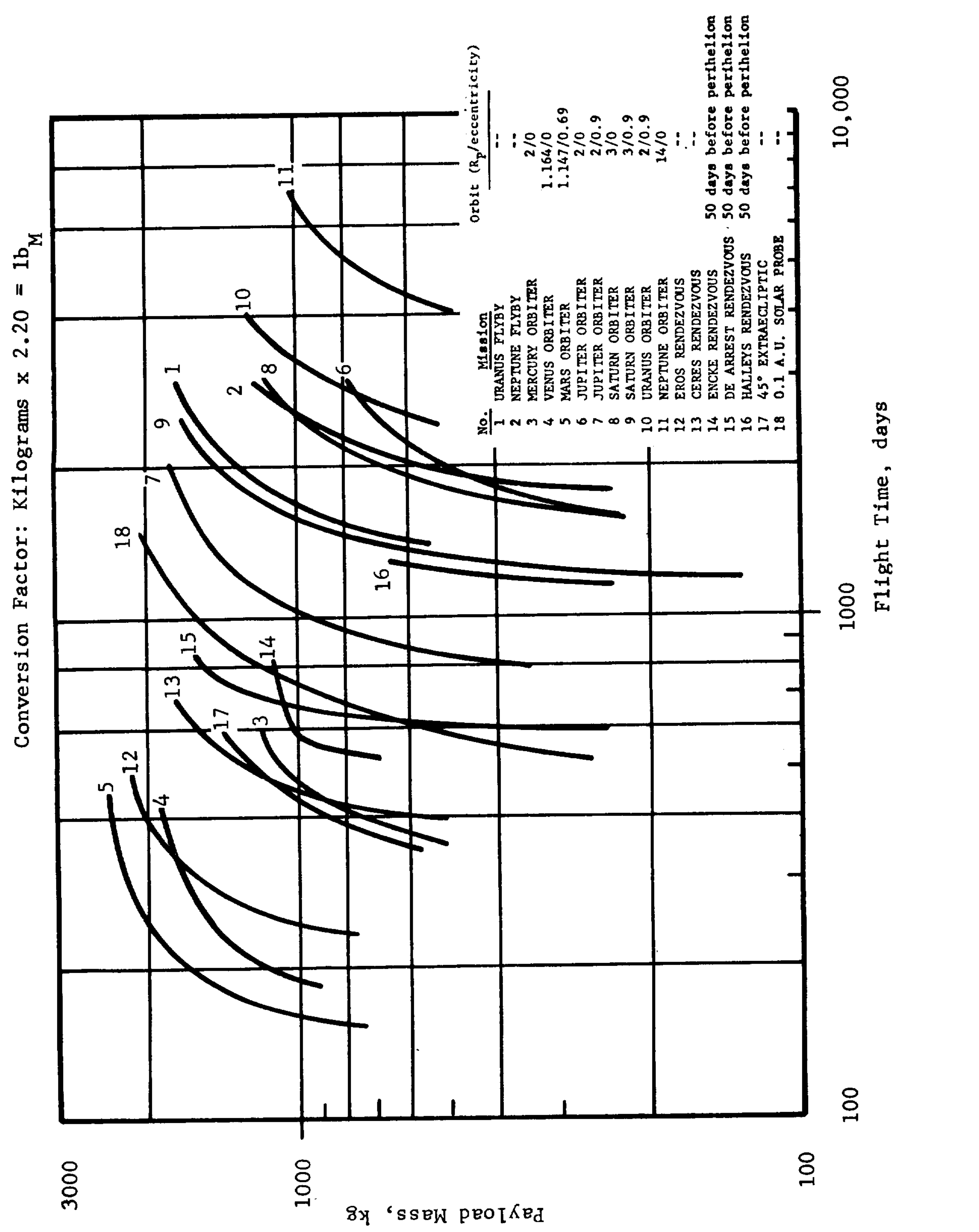


FIGURE 10-3. PERFORMANCE OF TITAN IIIE/CENTAUR/NEP (100 KWE)

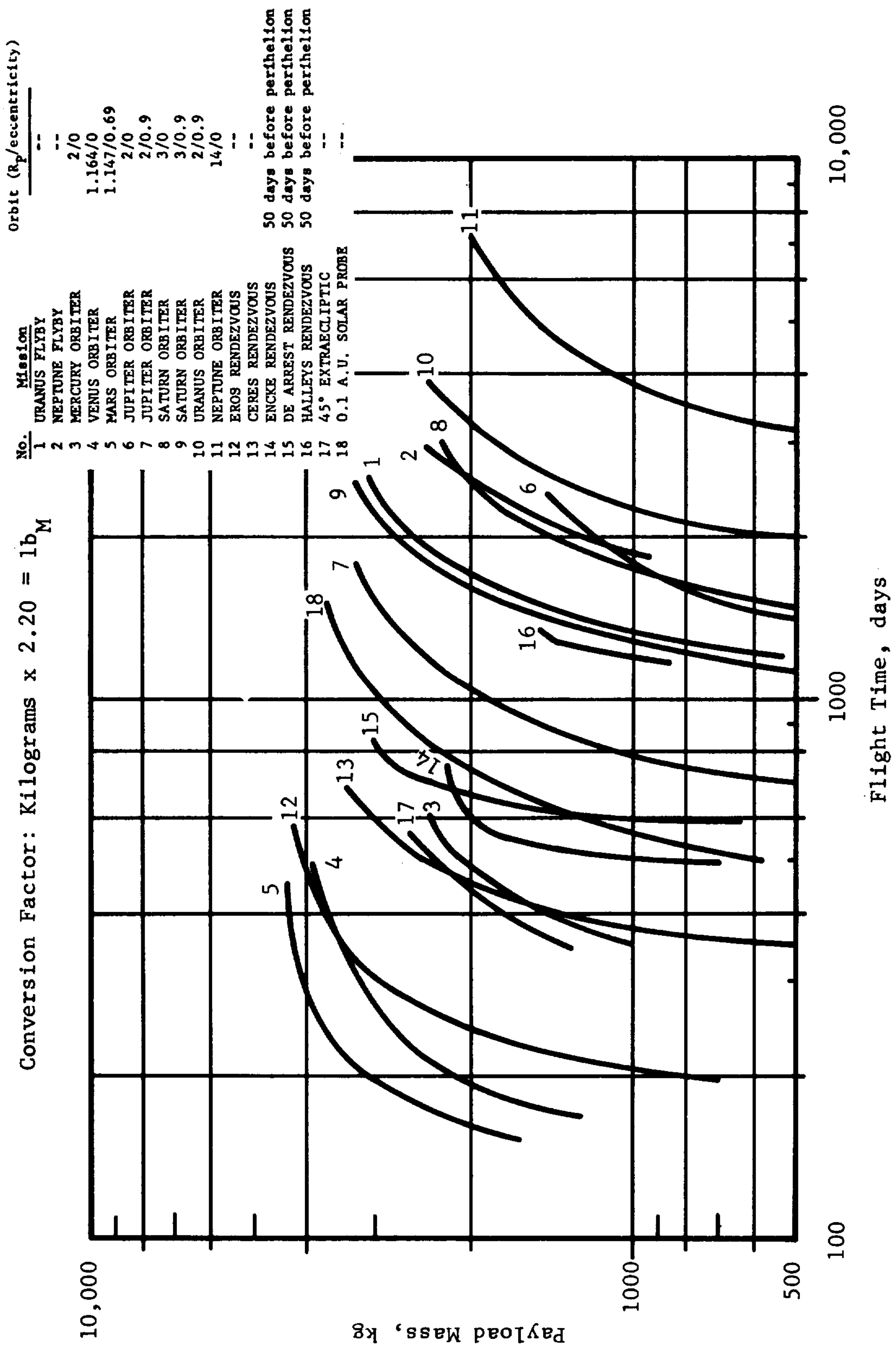


FIGURE 10-4. PERFORMANCE OF TITAN III7/CENTAUR/NEP (100 KWE)

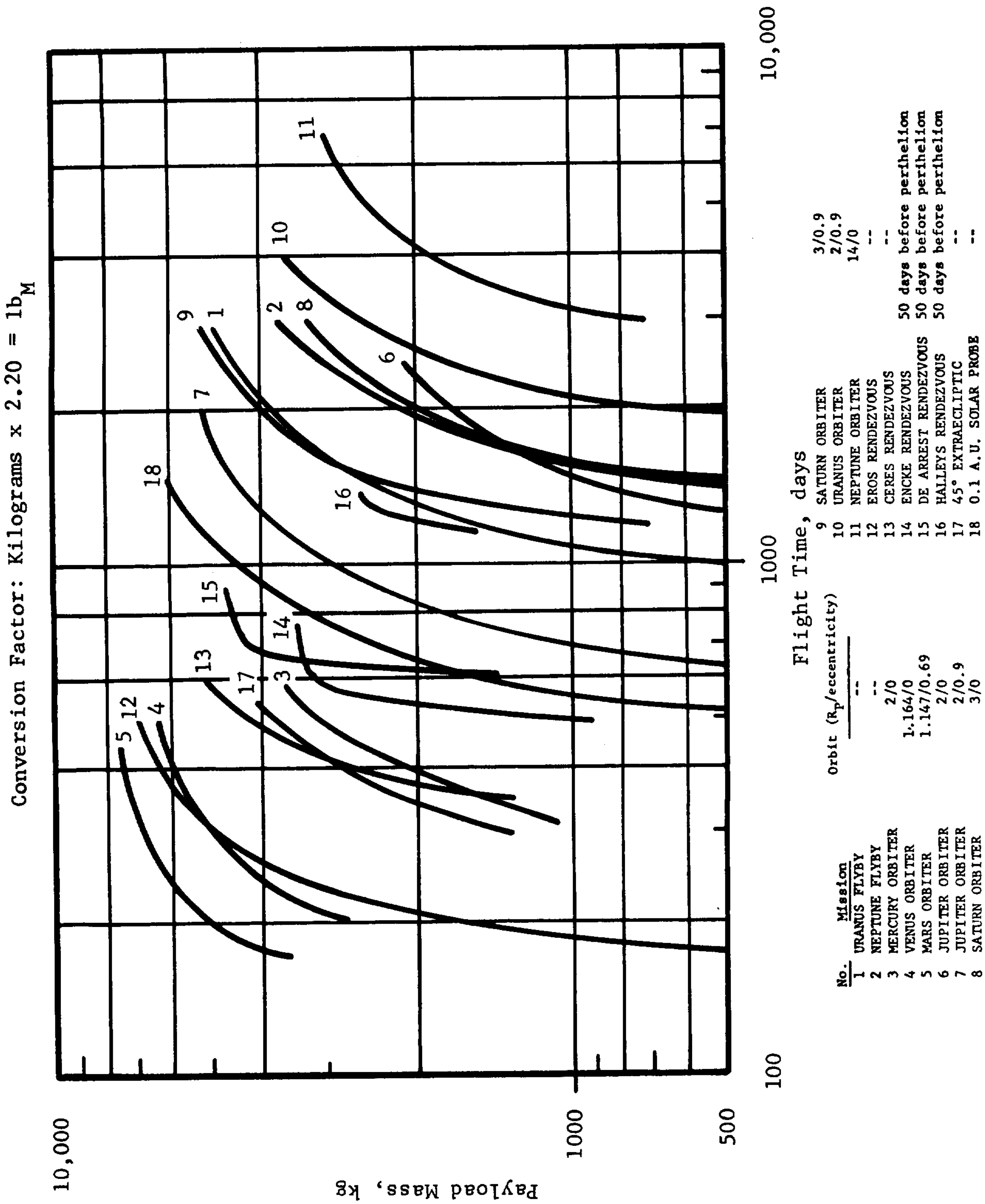
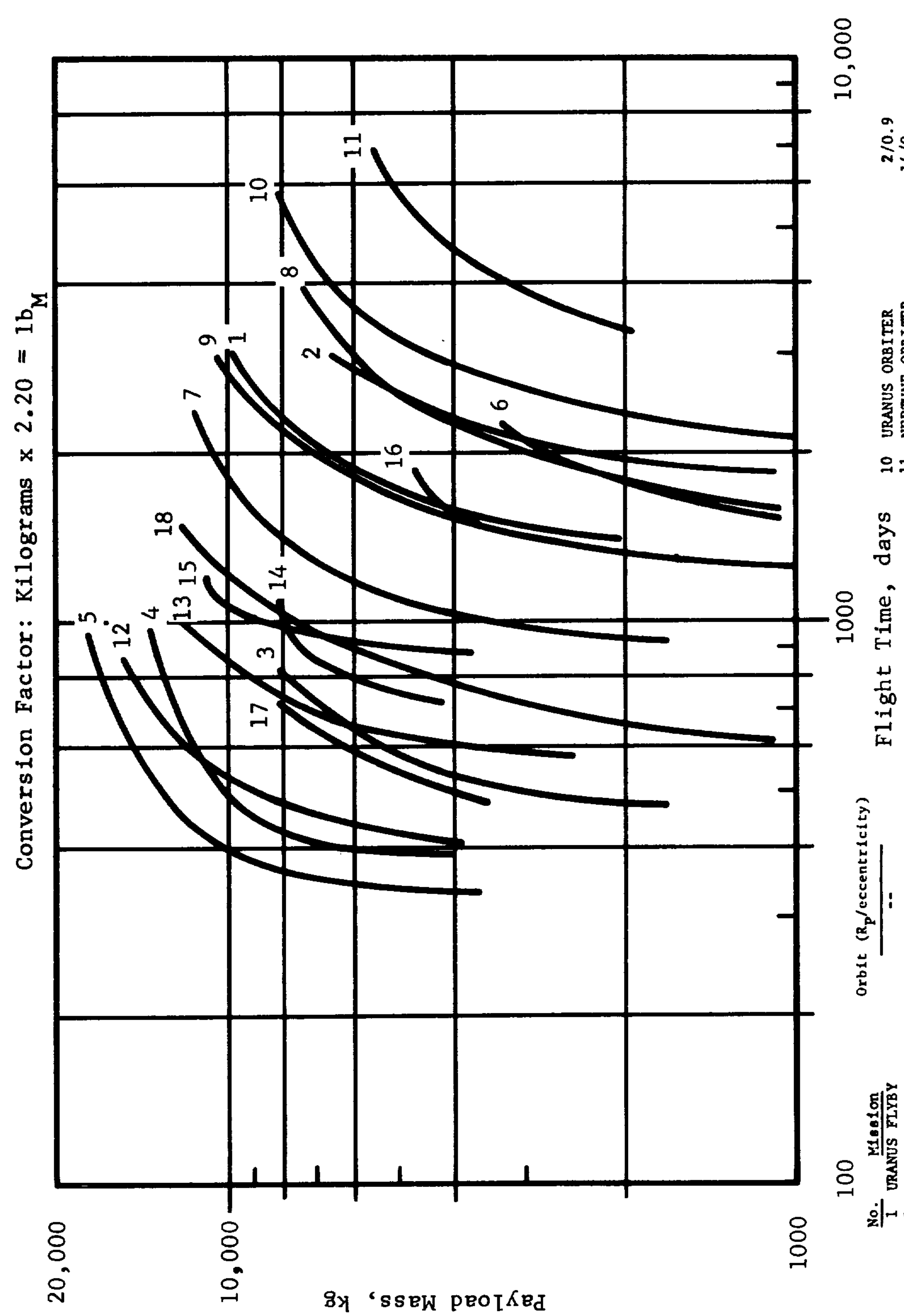


FIGURE 10-5. PERFORMANCE OF SPACE SHUTTLE (FULLY RECOVERABLE)/CENTAUR/NEP (100 KWE)



No.	Mission	Orbit (R_p/eccentricity)
1	URANUS FLYBY	--
2	NEPTUNE FLYBY	--
3	MERCURY ORBITER	2/0
4	VENUS ORBITER	1.164/0
5	MARS ORBITER	1.147/0.69
6	JUPITER ORBITER	2/0
7	JUPITER ORBITER	2/0.9
8	SATURN ORBITER	3/0
9	SATURN ORBITER	3/0.9
10	URANUS ORBITER	2/0.9
11	NEPTUNE ORBITER	14/0
12	EROS RENDEZVOUS	--
13	CERES RENDEZVOUS	--
14	ENCKE RENDEZVOUS	50 days before perihelion
15	DE ARREST RENDEZVOUS	50 days before perihelion
16	HALLEYS RENDEZVOUS	50 days before perihelion
17	45° EXTRAECLIPTIC	--
18	0.1 A.U. SOLAR PROBE	--

FIGURE 10-6. PERFORMANCE OF SPACE SHUTTLE (FULLY RECOVERABLE)/NEP (250 KWE)

CHAPTER 11: ESTIMATING FACTORS FOR VELOCITY IMPULSE MOTORS

TABLE OF CONTENTS

CHAPTER 11. ESTIMATING FACTORS FOR VELOCITY IMPULSE MOTORS

1100 GENERAL

1. This chapter provides information for making preliminary planning estimates of the mass of velocity impulse motors for such applications as retro-propulsion for entering planetary orbits and apogee-kick propulsion for establishing prescribed Earth orbits. Information concerning the estimation of velocity impulse requirements for various kinds of maneuvers is presented in Chapters 2 and 3.

2. Figures 11-1 through 11-4 show total propulsion system mass as a function of spacecraft payload (exclusive of the propulsion system) and the parameter $\Delta V/I$ where ΔV is the required velocity impulse (m/sec) and I is the specific impulse (m/sec). Specific impulse depends, among other factors, on the type of propellants to be used. Figures 11-1, 11-2, and 11-3 show typical ranges of the parameter $\Delta V/I$ for Earth-storable, mildly cryogenic space storable, and deep cryogenic liquid propellants, respectively. Figure 11-4 shows a typical range of the parameter for solid propellants.

3. Liquid propulsion systems employing Earth-storable propellants, such as N_2O_4/Aerozine 50, yield specific impulses in the range of 2800 to 3100 m/sec (Figure 11-1). The mildly cryogenic space-storable propellants, such as Flox/MMH or Flox/CH_4, produce specific impulses in the range of 3700 to 3900 m/sec (Figure 11-2). Deep cryogenic propellants, such as H_2/F_2, produce specific impulses in the range of 4450 to 4550 m/sec (Figure 11-3). The specific impulse may be assumed to be 2800 to 2850 m/sec for current solid propellant propulsion systems and up to 3050 m/sec for future solid propellant systems (Figure 11-4). For the estimating procedure in this chapter, the preceding values of specific impulse may be considered to represent conservative and optimistic limits.

4. The curves in Figures 11-1 through 11-4 are based on specific empirical relationships between size of a propulsion system and its mass fraction (ratio of propellant mass to total propulsion system mass). The mass fraction depends on the division of functions between the spacecraft and the propulsion system, the acceleration level chosen or permissible for the maneuver, and detailed mission-related factors. A thrust-to-propellant mass ratio of 10 m/sec^2 leads to

reasonable values of the burn time for the types of maneuvers being considered here. Some variation in the thrust-to-propellant mass ratio is permissible without significant changes in the data shown. The relationships used in Figures 11-1, 11-2, and 11-3 apply to liquid propulsion systems. It is assumed that the systems include a thrust vector control system and basic load-carrying structure, but do not include guidance, power, or control electronics systems. Propulsion system masses given in Figure 11-4 are for solid propellant motors where the spacecraft is assumed to provide all structural (other than motor case) and control functions; apogee-kick motors in spin-stabilizied spacecraft are examples of such motors

1101 PROCEDURE

1. Information discussed in paragraphs 204, 205, and 206 and in paragraphs 302, 303, and 304 can be used for estimating ΔV requirements for various types of maneuvers. In order to estimate the size of the propulsion system required to perform the maneuvers, it is first necessary to select a specific impulse value corresponding to the type of propellant and propulsion system to be used (refer to paragraph 1100. 3). Next, the value of the ratio $\Delta V/I$ is calculated for the selected ΔV. Then the propulsion system mass can be obtained for a specified spacecraft payload by using the appropriate figure in this chapter; interpolation between the curves of constant $\Delta V/I$ may be required.

2. If the required velocity increments are large, the use of two or more propulsion stages should be considered, especially if the total propulsion system mass becomes large compared to the spacecraft mass. Obviously, there is a trade off between system mass and system complexity. Multistage propulsion systems should be considered when the propulsion-system mass exceeds that of the spacecraft by a factor of about 10. This value is indicated by the dashed lines in the figures. As a first approximation, the velocity increment may be divided equally among the propulsion stages, using the minimum number of stages required to maintain the ratio of the final, or smallest stage mass to spacecraft mass below about 10. Then, each stage can be sized from the appropriate figure; it must be remembered that W_L for the lower stages must include the spacecraft mass plus that of all the upper stages as well

3. The propulsion system mass determined by the methods of this chapter should be added to the spacecraft mass to obtain the total payload for the launch vehicle. This total can then be used for the selection of a launch vehicle to perform the initial part of the mission (refer to Chapters 5, 6, 7, or 9).

1102 CAUTIONARY NOTE

Because of the many factors which influence the propellant choice and the design and performance of spacecraft propulsion systems, the propulsion system masses obtained by the procedures discussed in this chapter should be used only as estimates for planning purposes. Assistance may be obtained by contacting any of the persons listed in the Preface of this document.

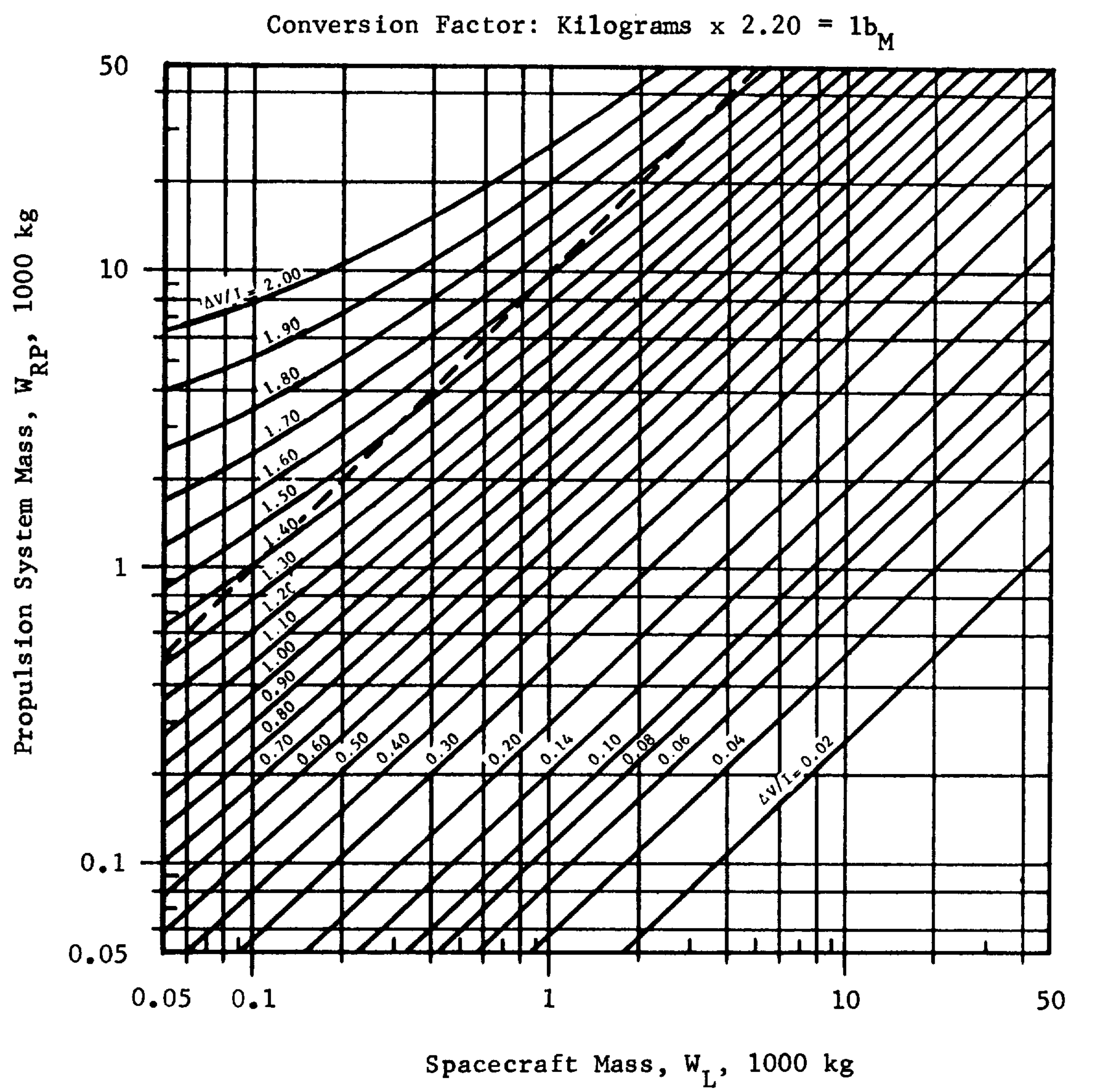


FIGURE 11-1. SPACECRAFT PROPULSION SYSTEM MASS USING EARTH-STORABLE PROPELLANTS

Notes:

1. Recommended for Earth orbital missions and planetary missions inward from Earth.
2. Electrical power, guidance and control electronics and telemetry systems are assumed to be included in spacecraft (not included in W_{RP}).
3. Calculations based on pressure fed engines and thrust = propellant weight.
4. Dashed line indicates $W_{RP}/W_L = 10$.
5. I is specific impulse in meters per second.

Conversion Factor: Kilograms x 2.20 = lb_M

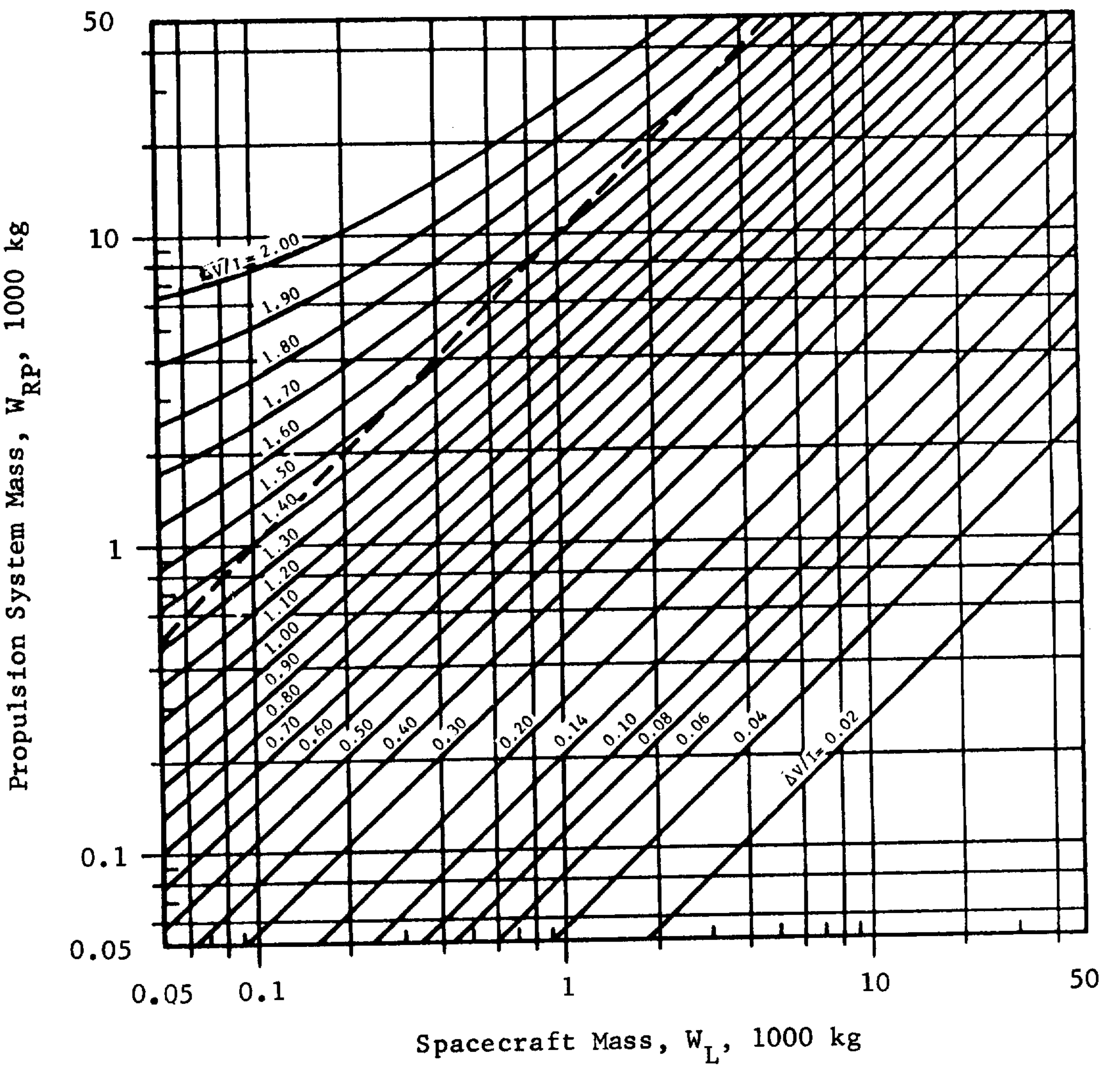


FIGURE 11-2. SPACECRAFT PROPULSION SYSTEM MASS USING SPACE-STORABLE PROPELLANTS

Notes:

1. Recommended for Earth orbital missions and planetary missions outward from Earth.
2. Electrical power, guidance and control electronics and telemetry systems are assumed to be included in spacecraft (not included in W_{RP}).
3. Calculations based on pressure fed engines and thrust = propellant weight.
4. Dashed line indicates W_{RP}/W_L = 10.
5. I is specific impulse in meters per second.

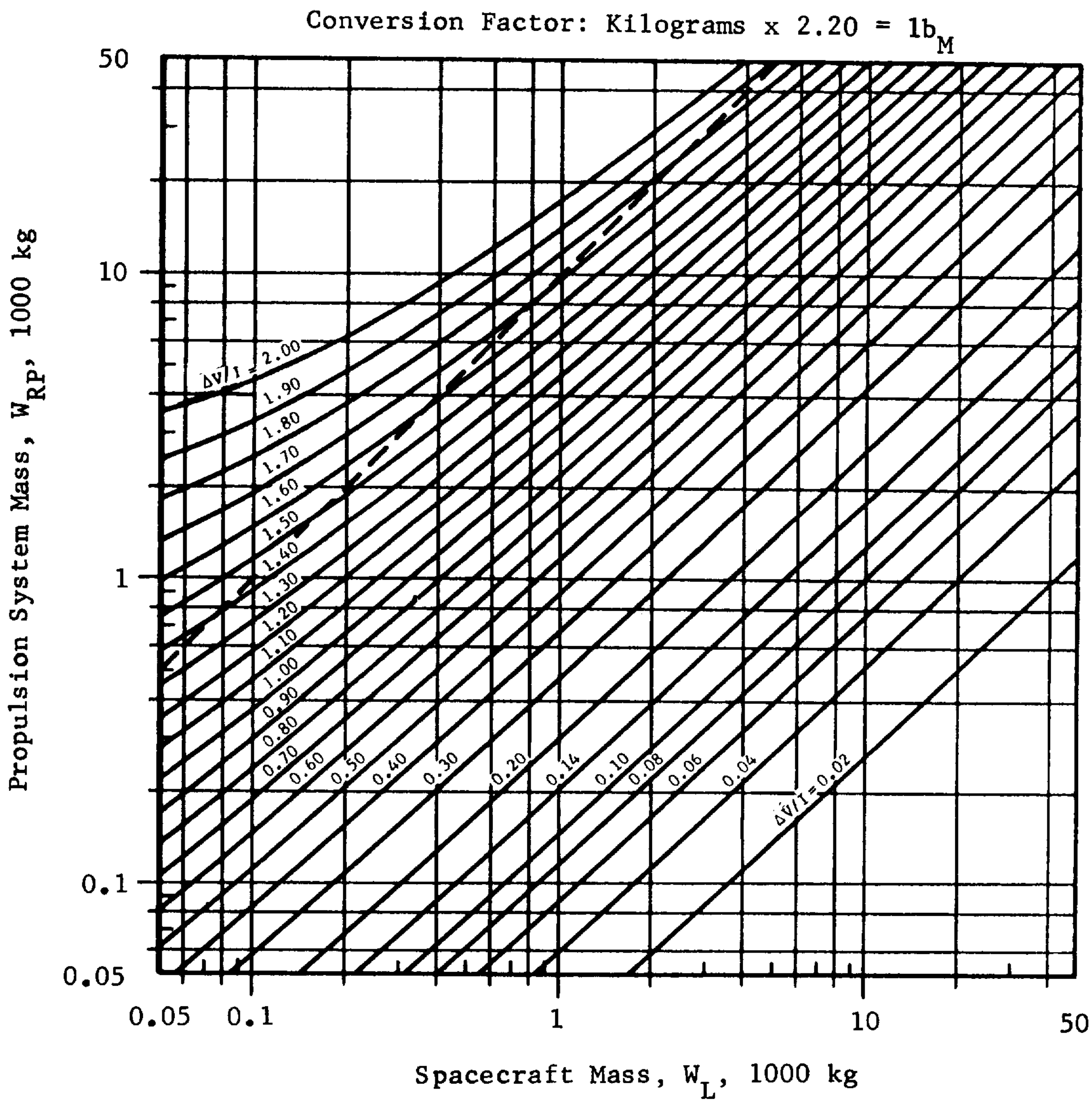


FIGURE 11-3. SPACECRAFT PROPULSION SYSTEM MASS USING CRYOGENIC PROPELLANTS

Notes:

1. Recommended for Earth orbital missions and planetary missions outward from Earth.
2. Electrical power, guidance and control electronics and telemetry systems are assumed to be included in spacecraft (not included in W_{RP}).
3. Calculations based on pump fed engines and thrust = propellant weight.
4. Dashed line indicates $W_{RP}/W_L = 10$.
5. I is specific impulse in meters per second.

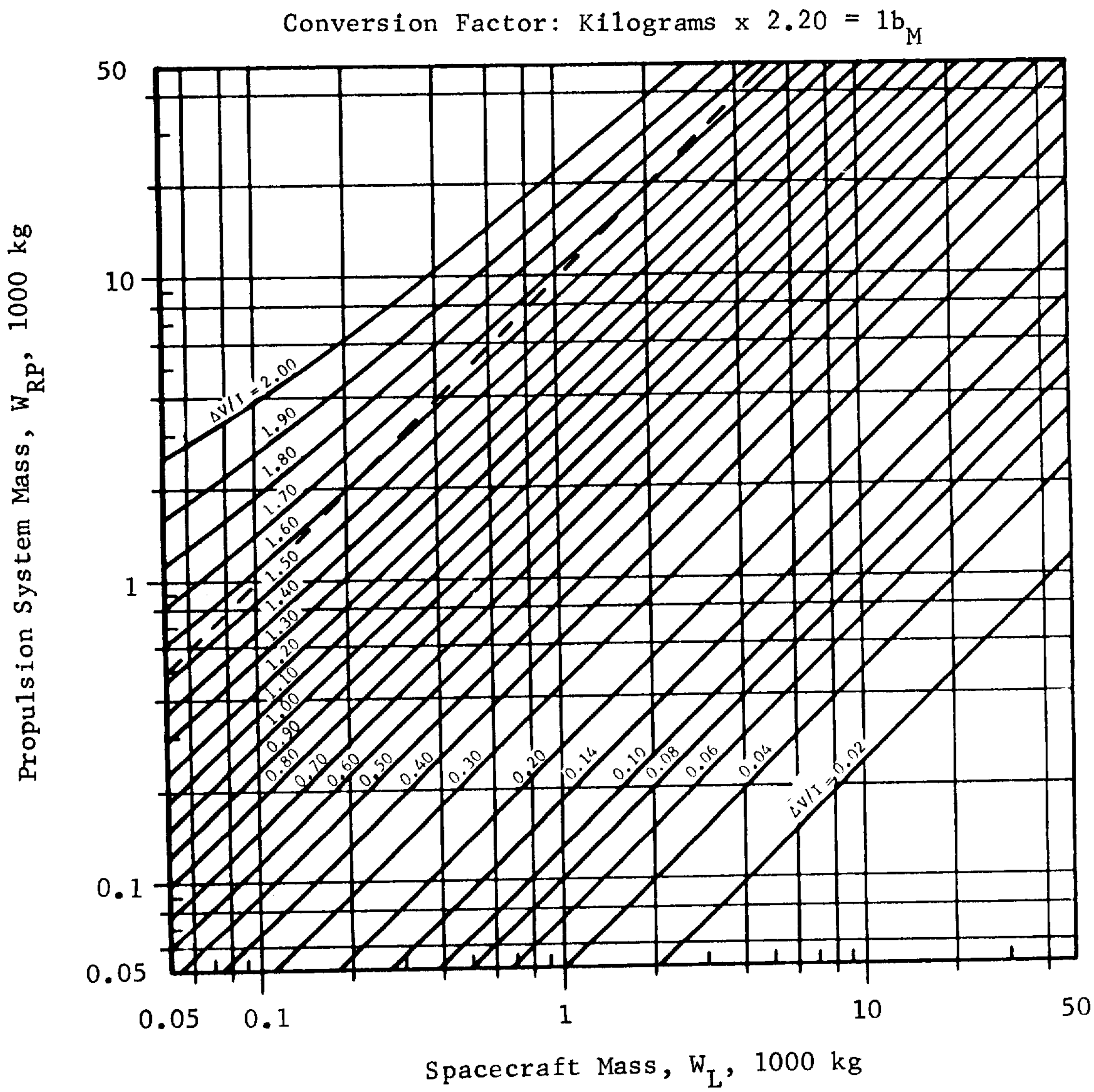


FIGURE 11-4. SPACECRAFT PROPULSION SYSTEM MASS USING SOLID PROPELLANTS

Notes:

1. Recommended for apogee kick motors and Earth orbital missions.
2. Electrical power, guidance and control electronics and telemetry systems are assumed to be included in spacecraft (not included in W_{RP}).
3. Dashed line indicates $W_{RP}/W_L = 10$.
4. I is specific impulse in meters per second.

CHAPTER 12: SHROUD CONFIGURATIONS

TABLE OF CONTENTS

CHAPTER 12: SHROUD CONFIGURATIONS

1200 GENERAL

1. This chapter presents simplified line drawings of nominal shrouds for some launch vehicles and upper stages. Nominal dimensions and payload volumes, given in metric units, are indicated to provide guidelines for payload sizing considerations if nominal shrouds are to be used.

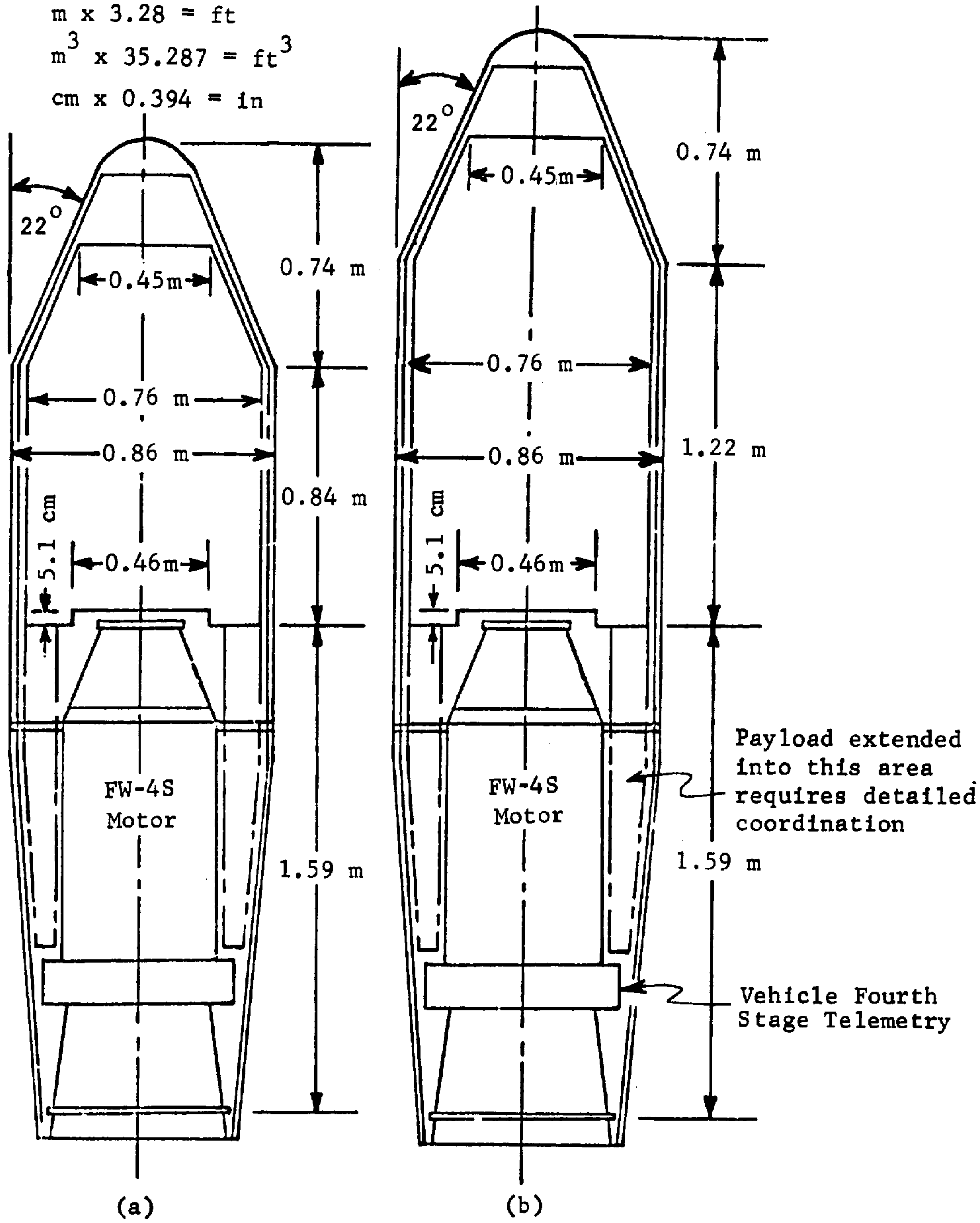
2. The size and shape of a proposed payload is an important consideration in advance mission planning. In some cases, these factors may be as important as payload mass and characteristic velocity in selecting the appropriate launch vehicle for a given mission. If nominal shroud configurations are unsuitable, some modifications to the shrouds may be possible. Substantial modifications such as hammerheading and excessive lengthening, however, may require considerable analysis to determine feasibility. Correspondingly, such modification may be expensive in terms of increased development and recurring costs and may be detrimental in terms of launch vehicle performance. Questions concerning special shroud configurations should be addressed to the persons listed in the Preface.

Conversion factors:

m x 3.28 = ft

m^3 x 35.287 = ft^3

cm x 0.394 = in



Payload
Volume = 0.48 cubic meters

Payload
Volume = 0.65 cubic meters

FIGURE 12-1. 86-CM (34-INCH)-DIAMETER SCOUT SHROUDS

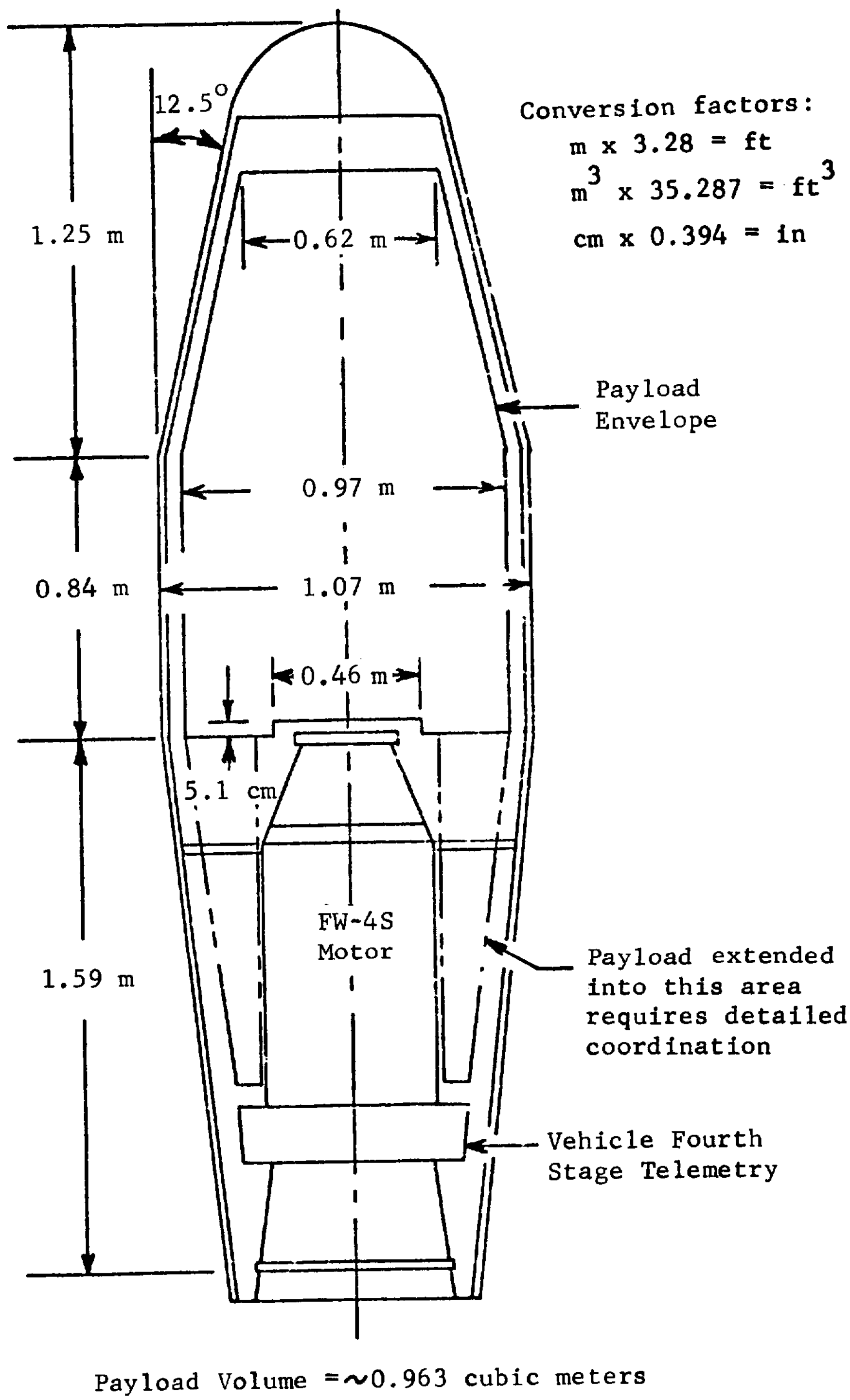


FIGURE 12-2. 1.07-M (42-INCH)-DIAMETER SCOUT SHROUD

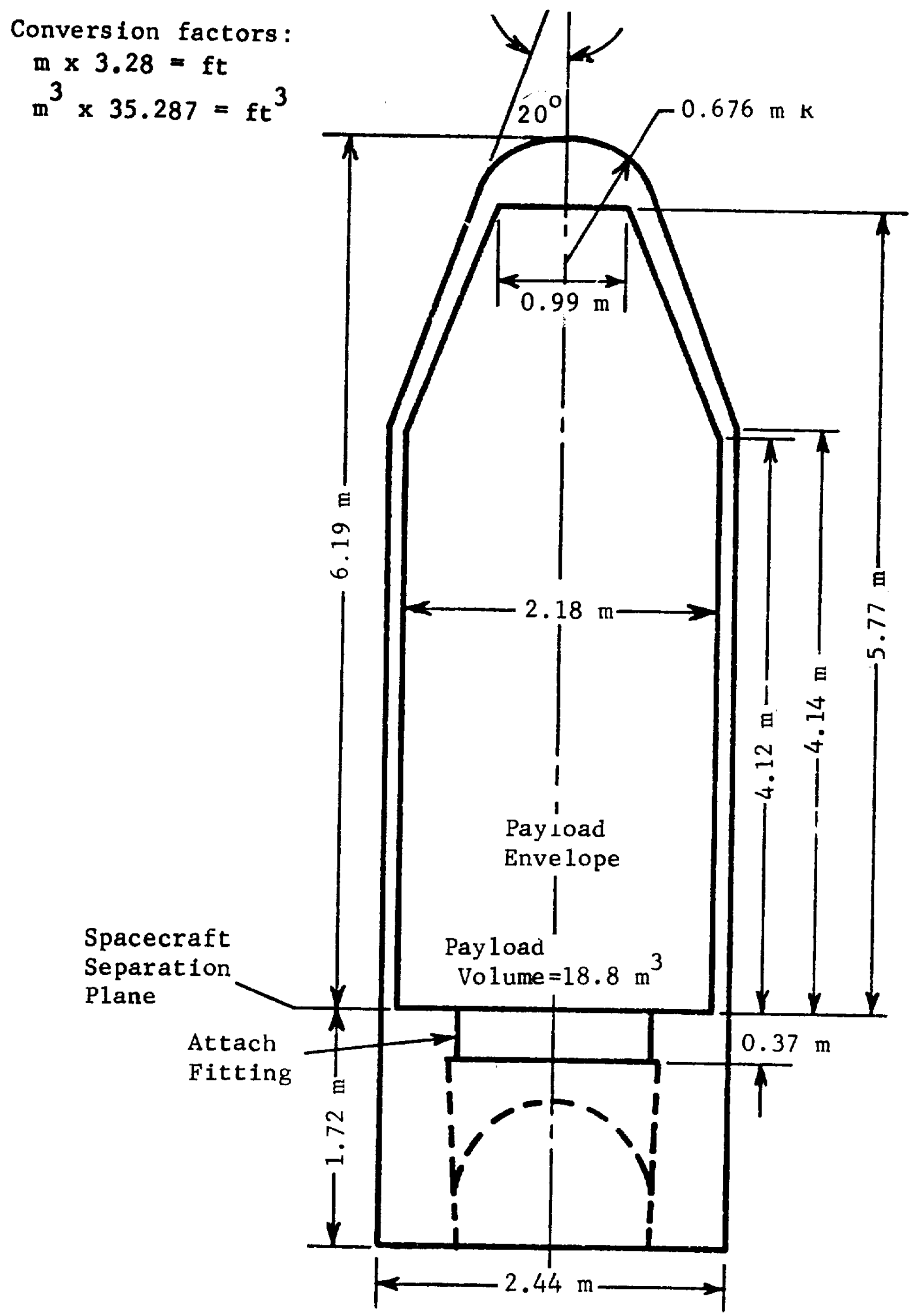


FIGURE 12-3. DELTA "STRAIGHT-8" FAIRING/SPACECRAFT ENVELOPE DIRECT MOUNTING ON SECOND STAGE

Conversion factors:
m x 3.28 = ft
m^3 x 35.287 = ft^3
cm x 0.394 = in

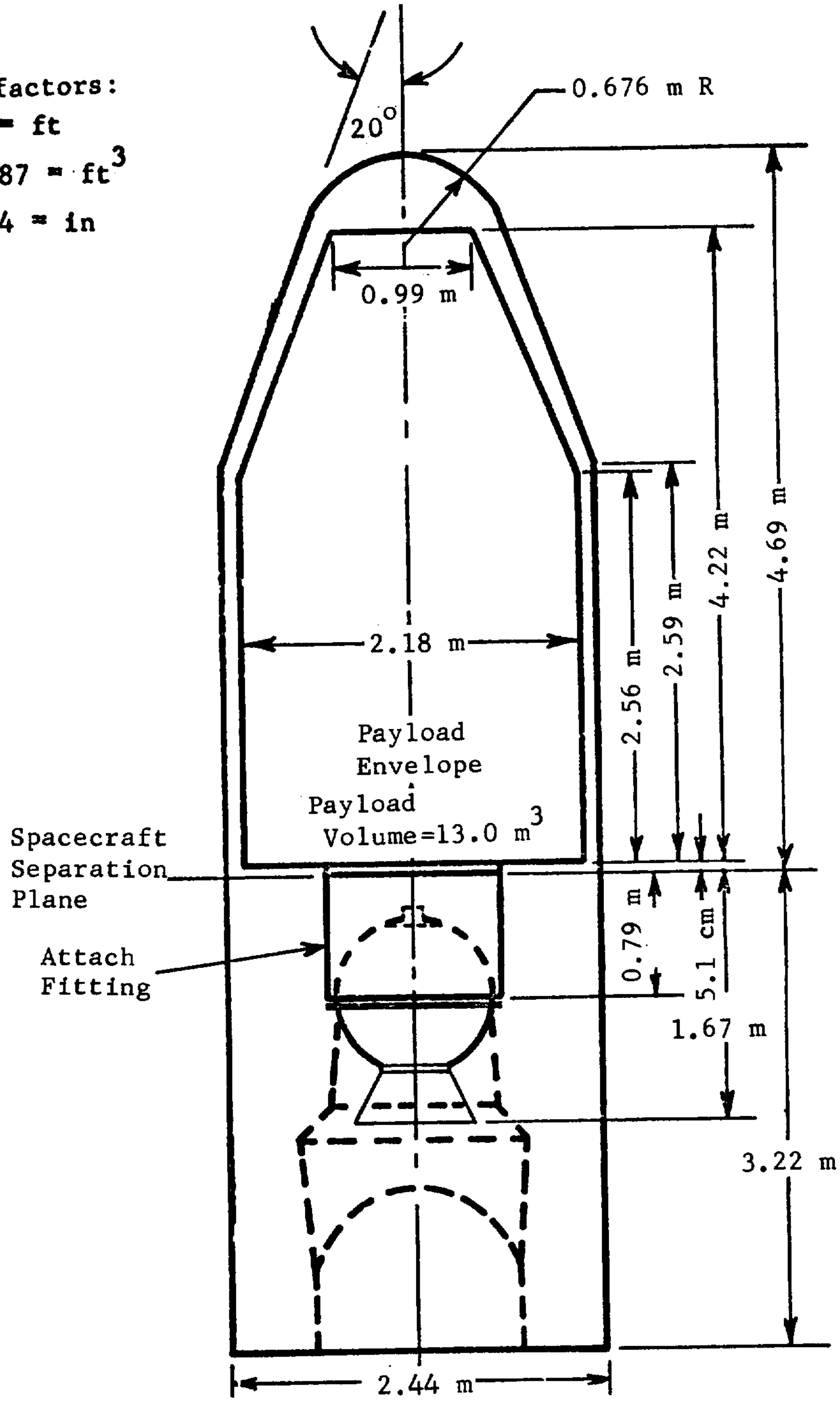


FIGURE 12-4. DELTA "STRAIGHT-8" FAIRING/SPACECRAFT ENVELOPE WITH THE TE364-3(1440) THIRD-STAGE MOTOR

Conversion factors:

m x 3.28 = ft

m^3 x 35.287 = ft^3

cm x 0.394 = in

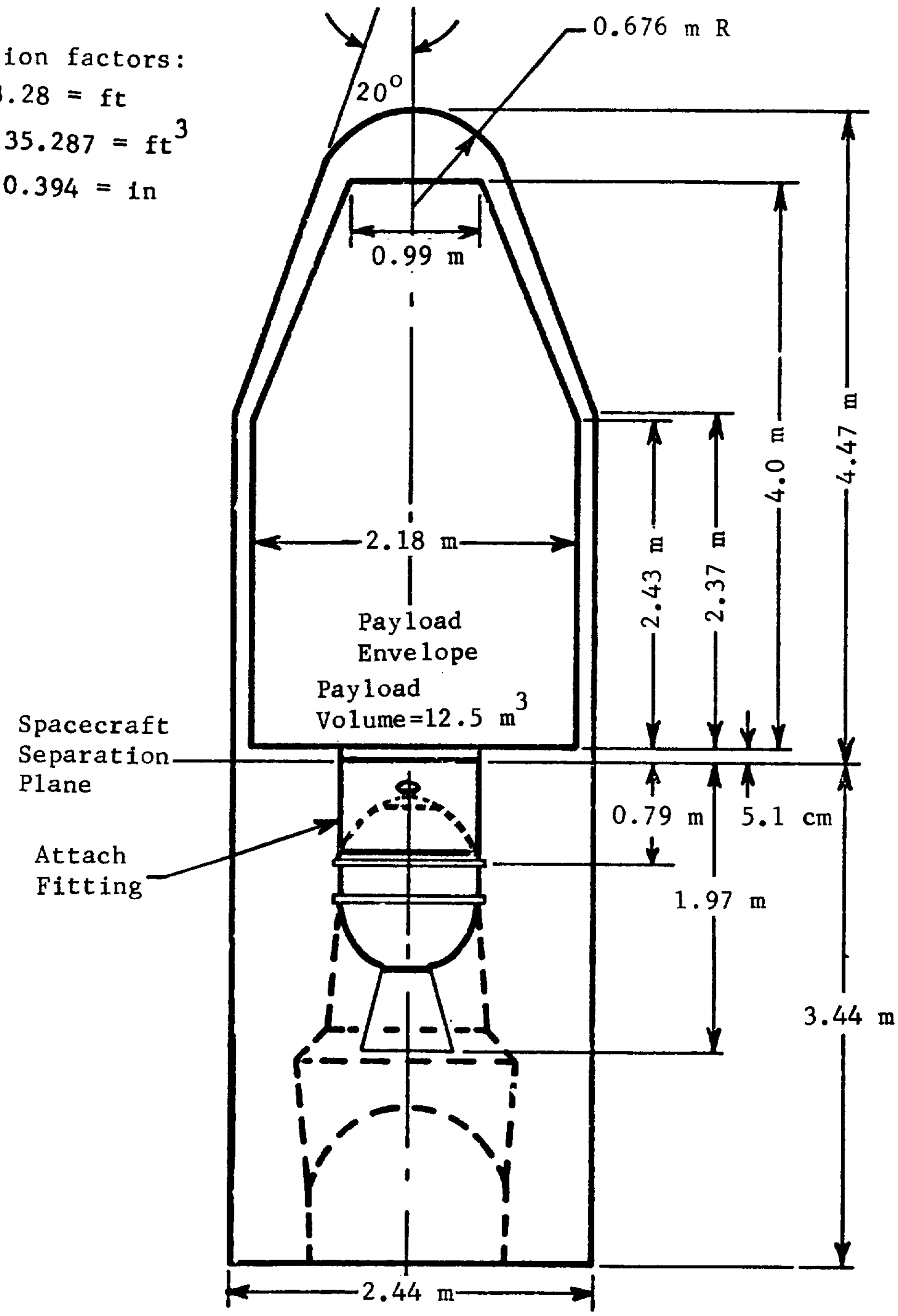


FIGURE 12-5. DELTA "STRAIGHT-8" FAIRING/SPACECRAFT ENVELOPE WITH THE TE364-4(2300) THIRD-STAGE MOTOR

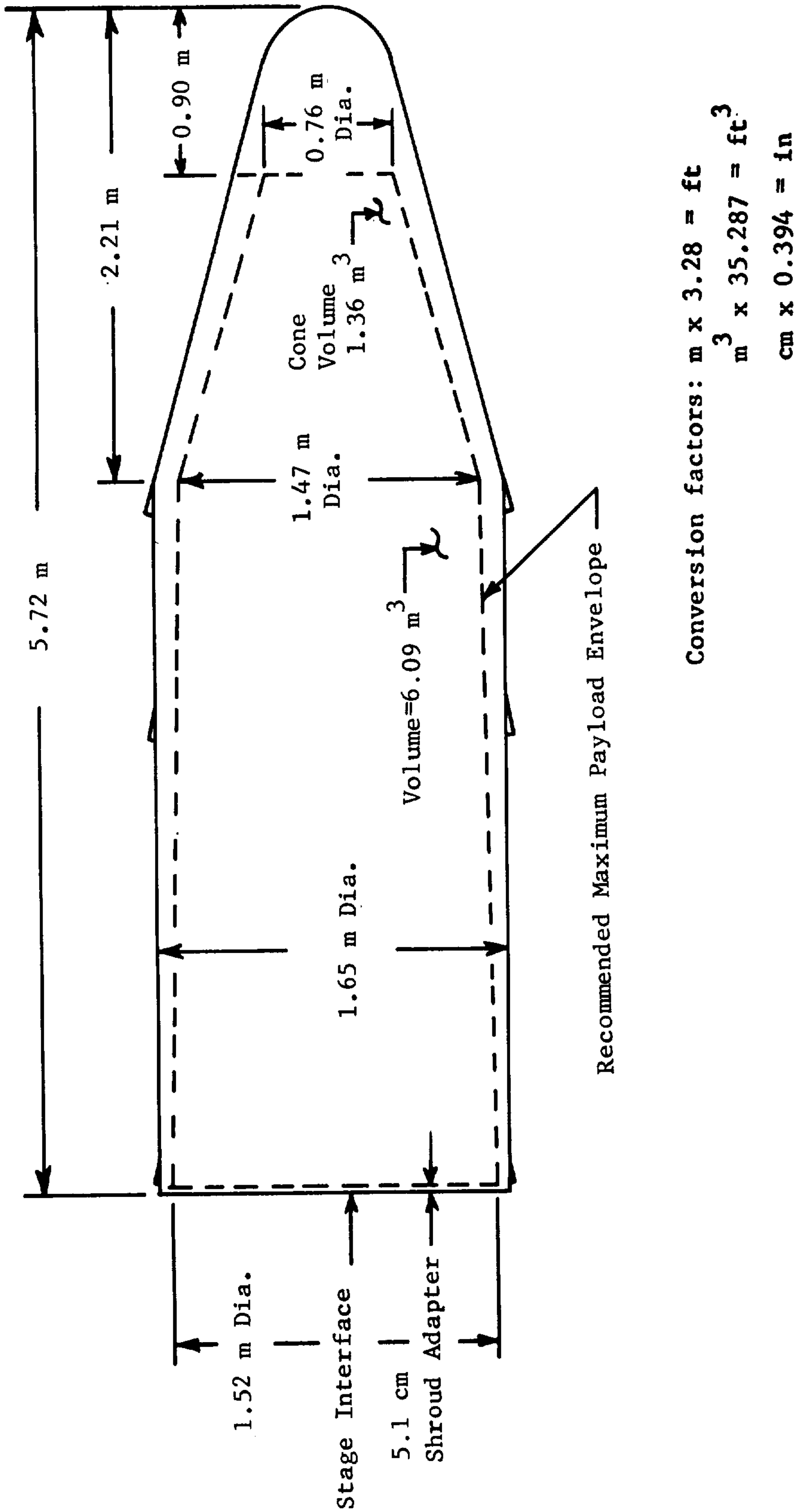


FIGURE 12-6. STANDARD AGENA CLAMSHELL SHROUD

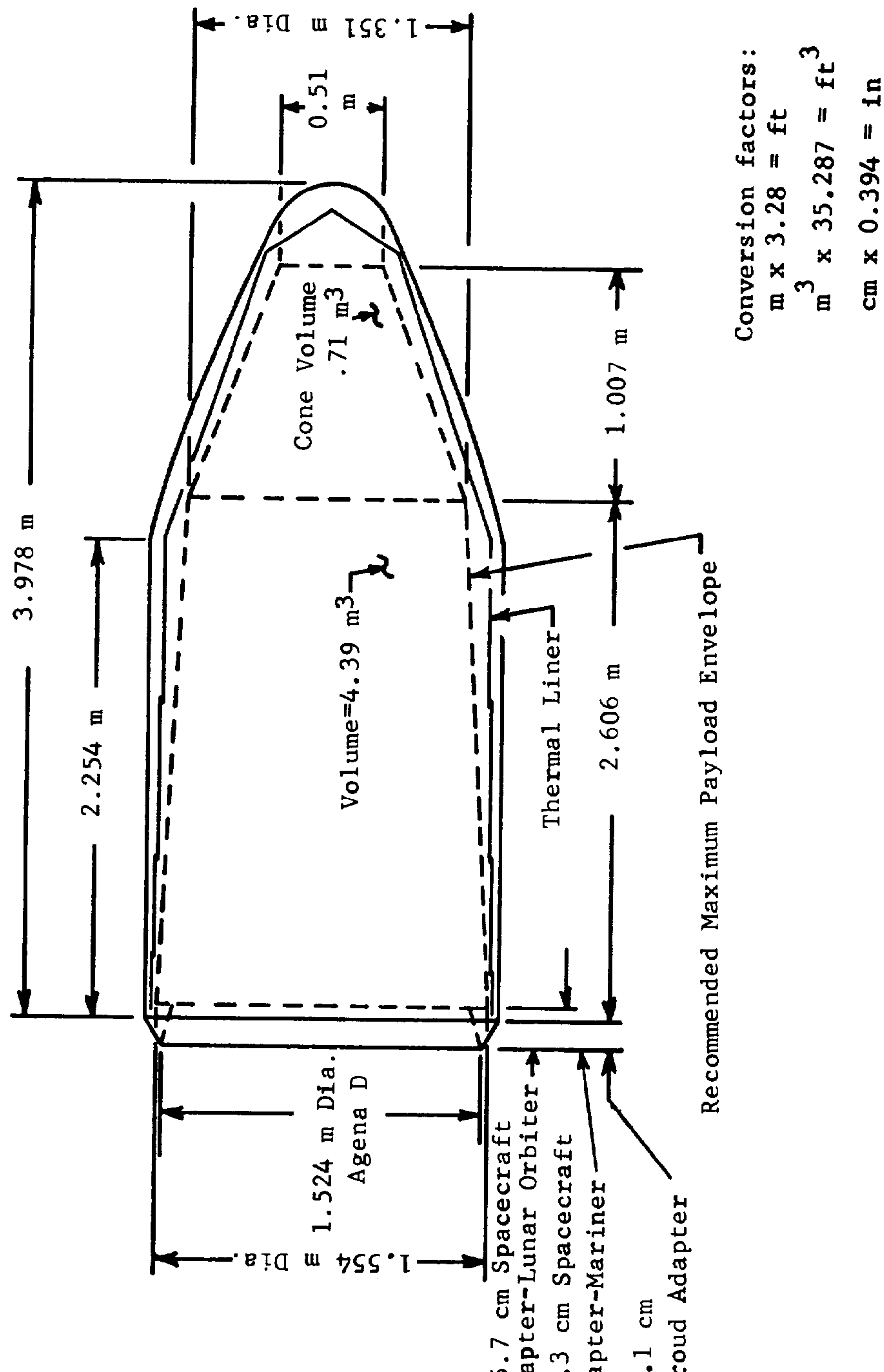


FIGURE 12-7. AGENA MARINER C AND LUNAR ORBITER OVER-THE-NOSE SHROUD

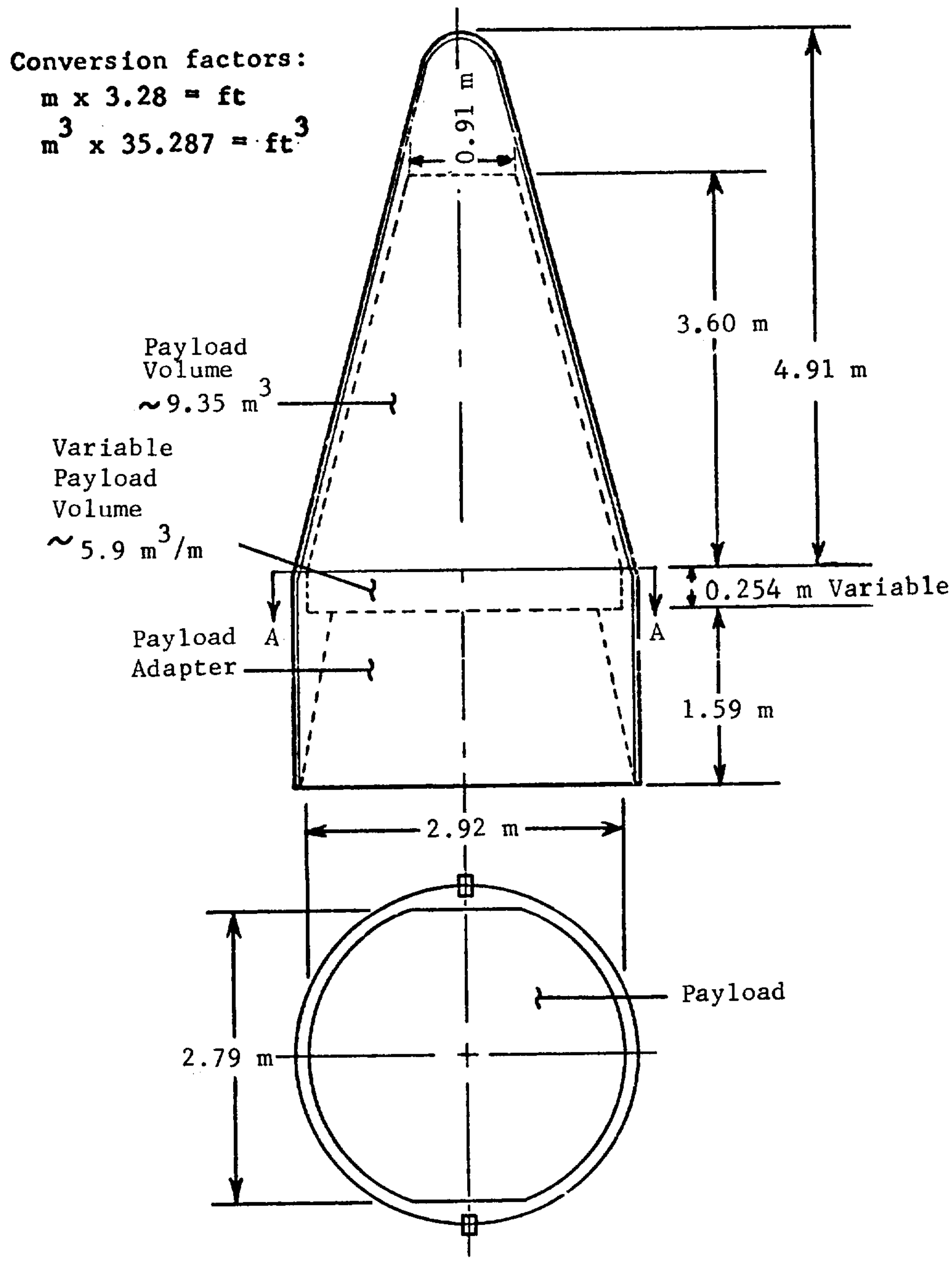


FIGURE 12-8. BASIC CENTAUR NOSE FAIRING

Conversion Factors: m x 3.28 = ft
m^3 x 35.287 = ft^3

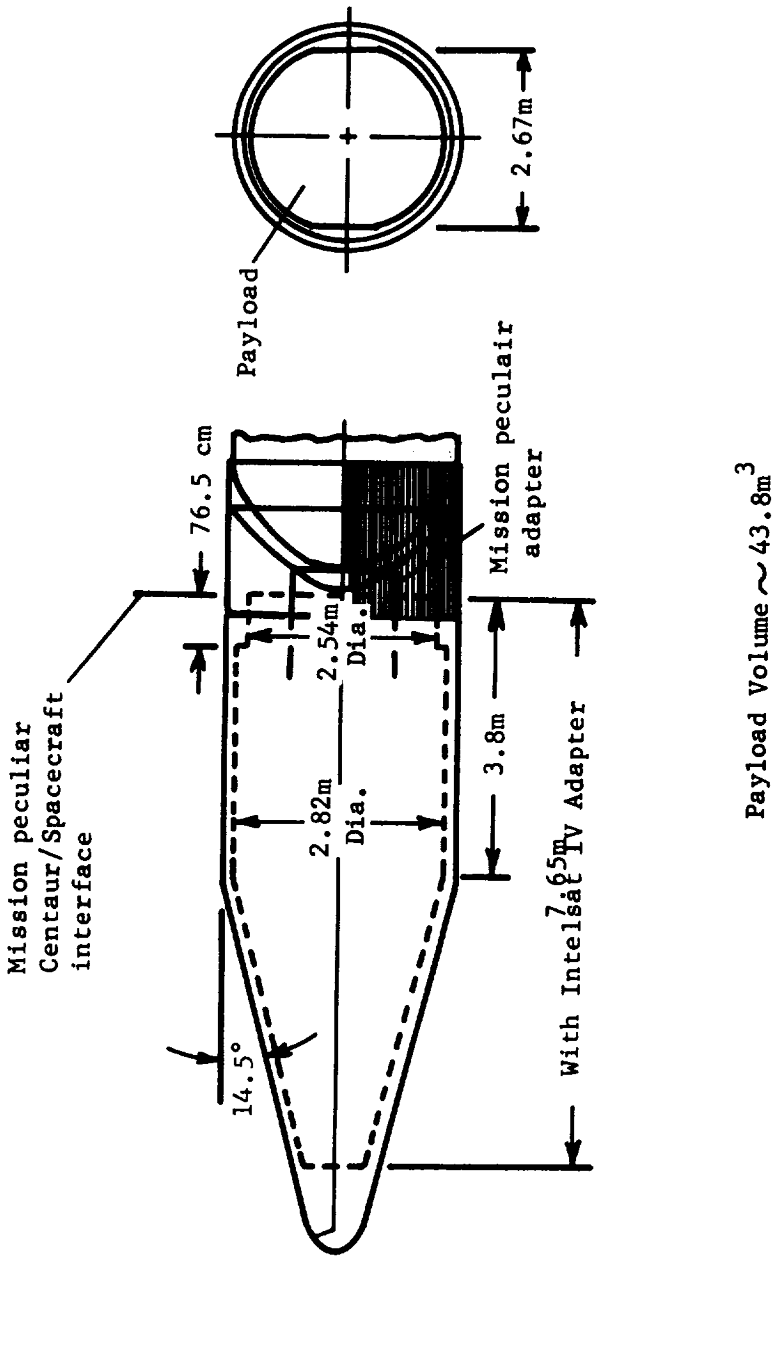
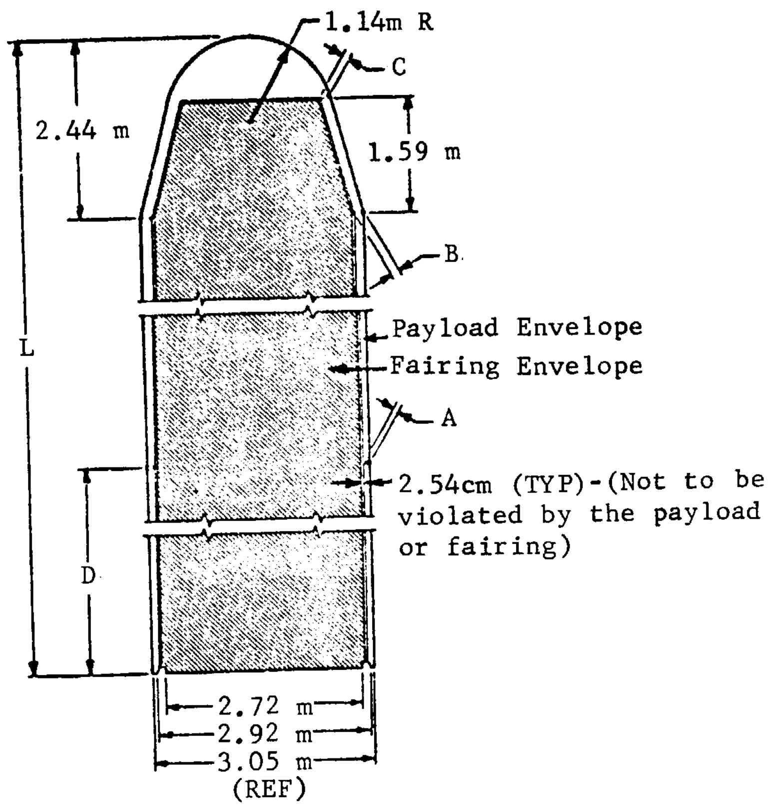


FIGURE 12-9. CENTAUR D-1A SHROUD

WEIGHT kg	CONFIG. NO., ft	L m	A cm	B cm	C cm	D m
618.7	XX15	4.57	7.62	7.62	7.62	2.13
756.6	XX20	6.10	7.62	7.62	7.62	3.66
892.2	XX25	7.62	7.62	7.62	7.62	3.66
1029.7	XX30	9.14	7.62	7.87	9.14	3.66
1169.4	XX35	10.67	7.62	9.40	10.7	3.66
1305.0	XX40	12.19	7.62	10.9	12.2	3.66
1442.0	XX45	13.72	7.62	12.4	13.7	3.66
1577.6	XX50	15.24	7.62	14.0	15.2	3.66

Conversion factors: m x 3.28 = ft
cm x 0.394 = in
kg x 2.20 = lb



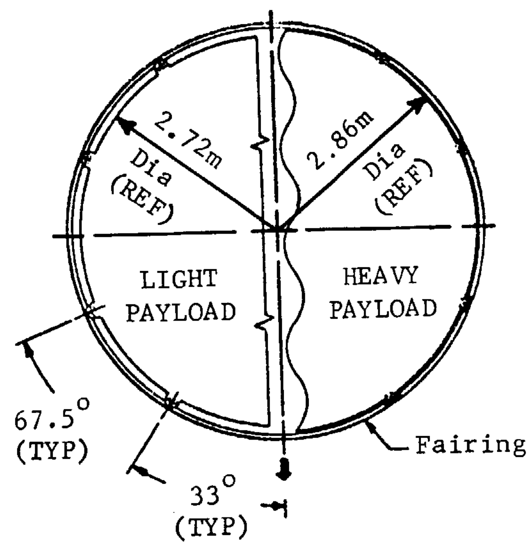
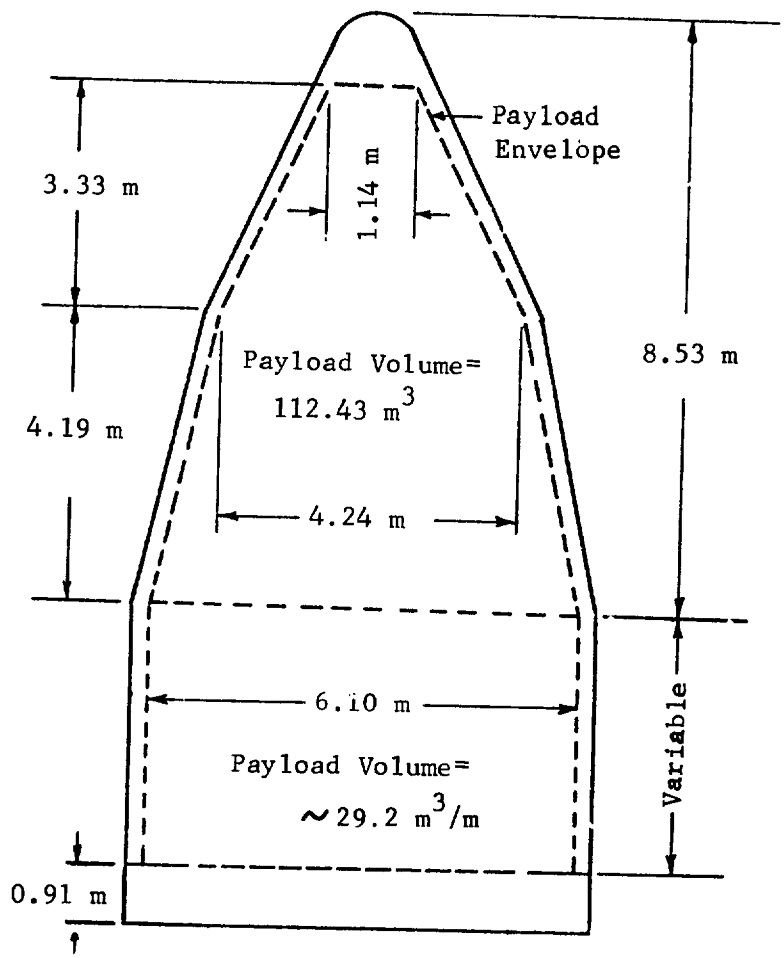


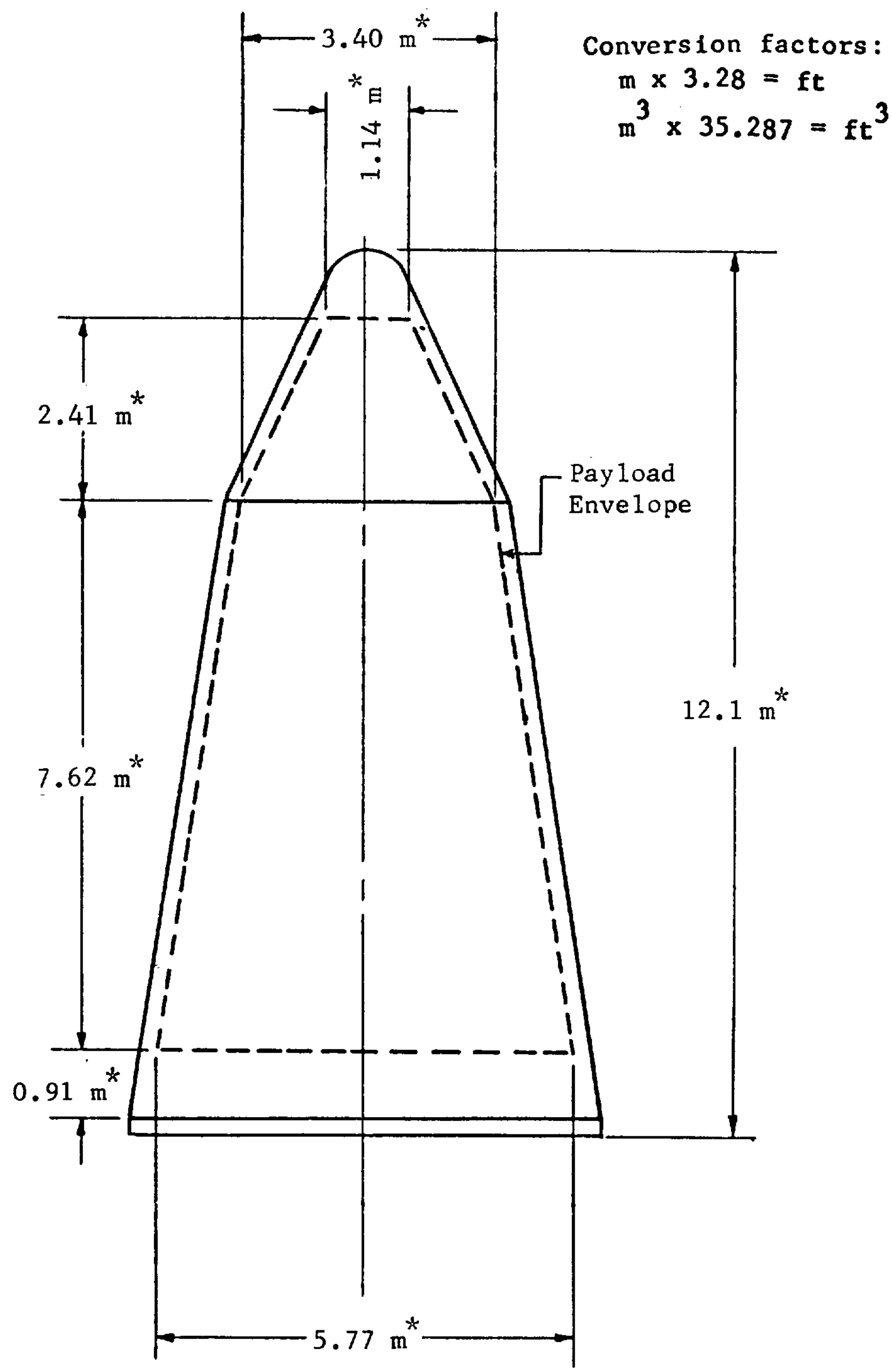
FIGURE 12-10. TITAN UPLF (UNIVERSAL PAYLOAD FAIRING)

Conversion factors: m x 3.28 = ft
m^3 x 35.287 = ft^3



Note: All dimensions to be used for preliminary layout only

FIGURE 12-11. SATURN IB AND SATURN V SHROUD (BASED ON VOYAGER NOSE CONE)

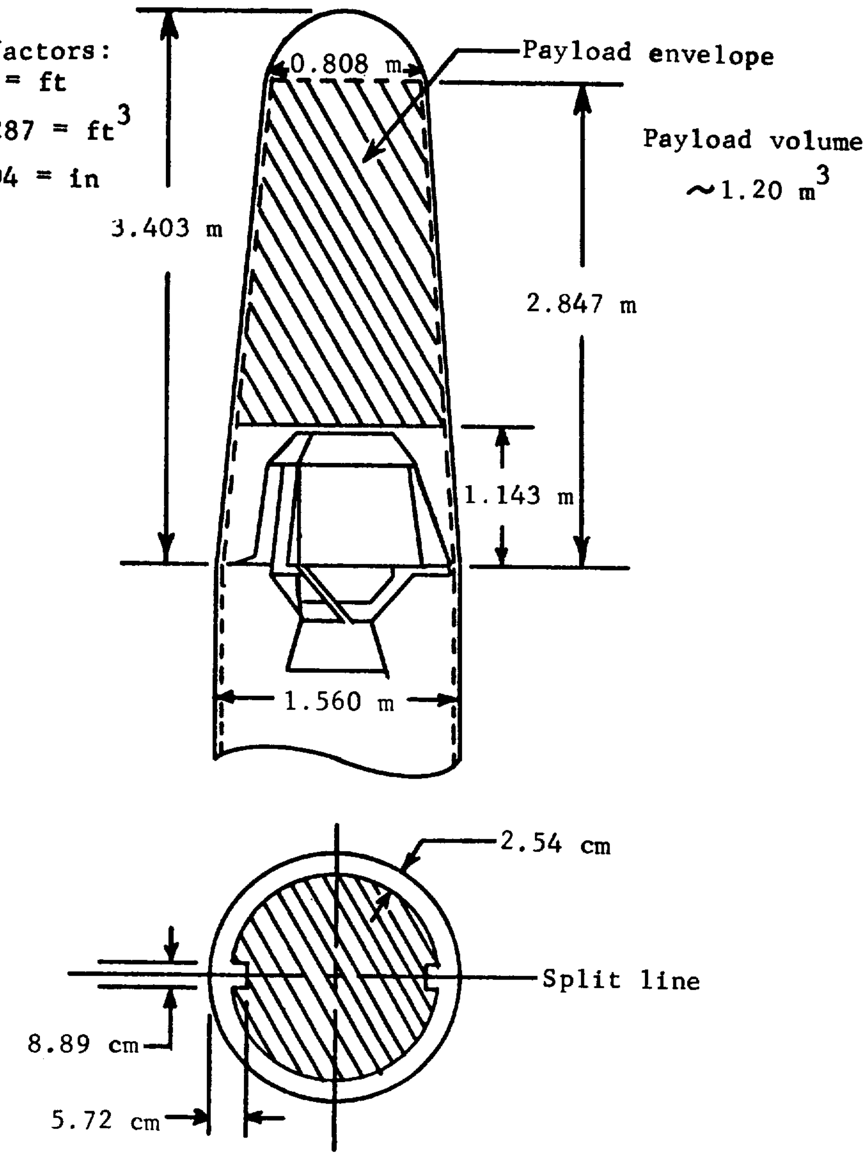


Basic Payload Volume is ~141.6 m^3

* Approximate dimensions to be used for preliminary layout only

FIGURE 12-12. SATURN IB AND SATURN V SHROUD (BASED ON MODIFIED LEM ADAPTER)

Conversion factors:
m x 3.28 = ft
m^3 x 35.287 = ft^3
cm x 0.394 = in



Typical cross section

Note: This fairing is being extended by the addition of a 5.76 m cylindrical section.

FIGURE 12-13. STANDARD BURNER II FAIRING SYSTEM

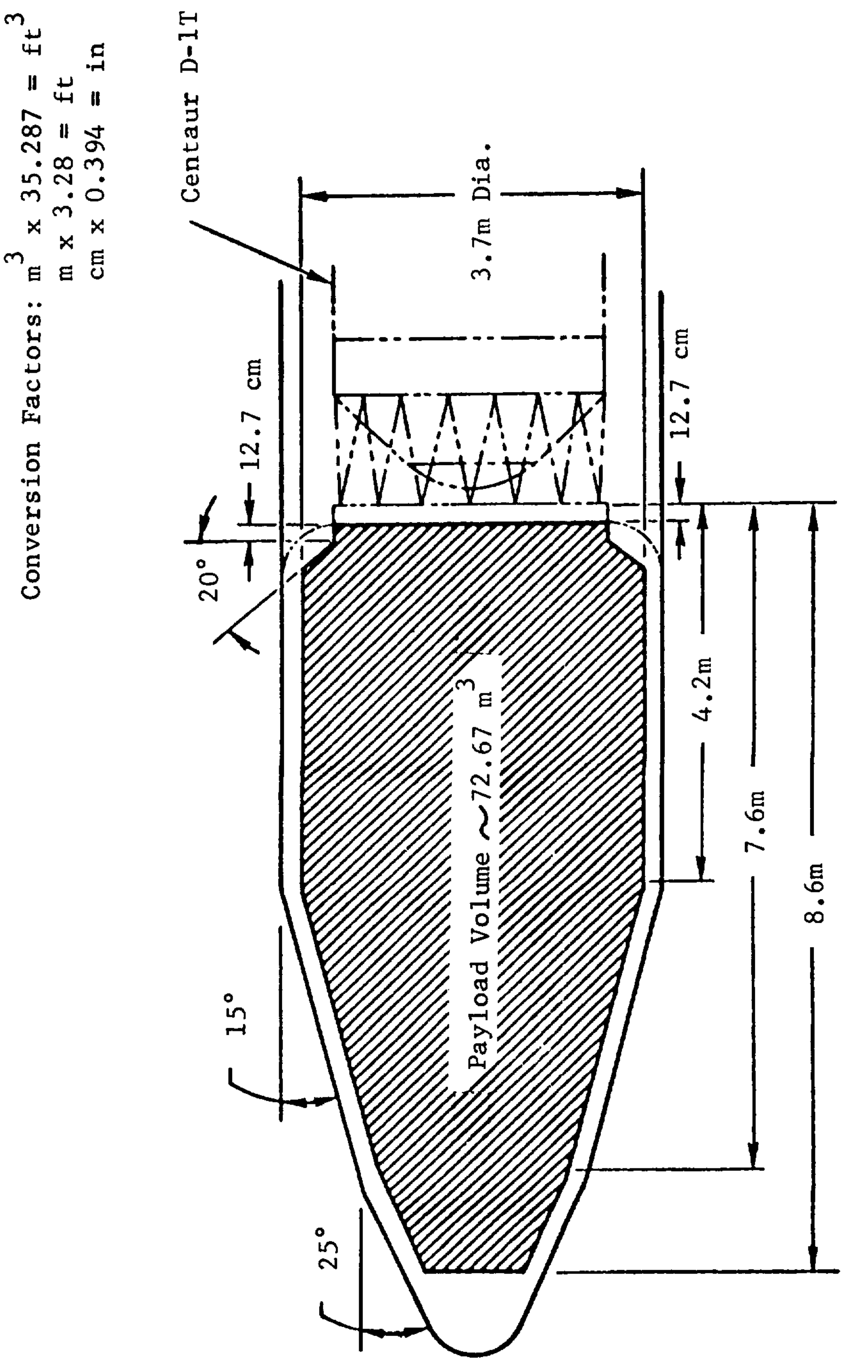


FIGURE 12-14. TITAN/CENTAUR STANDARD SHROUD PAYLOAD ENVELOPE

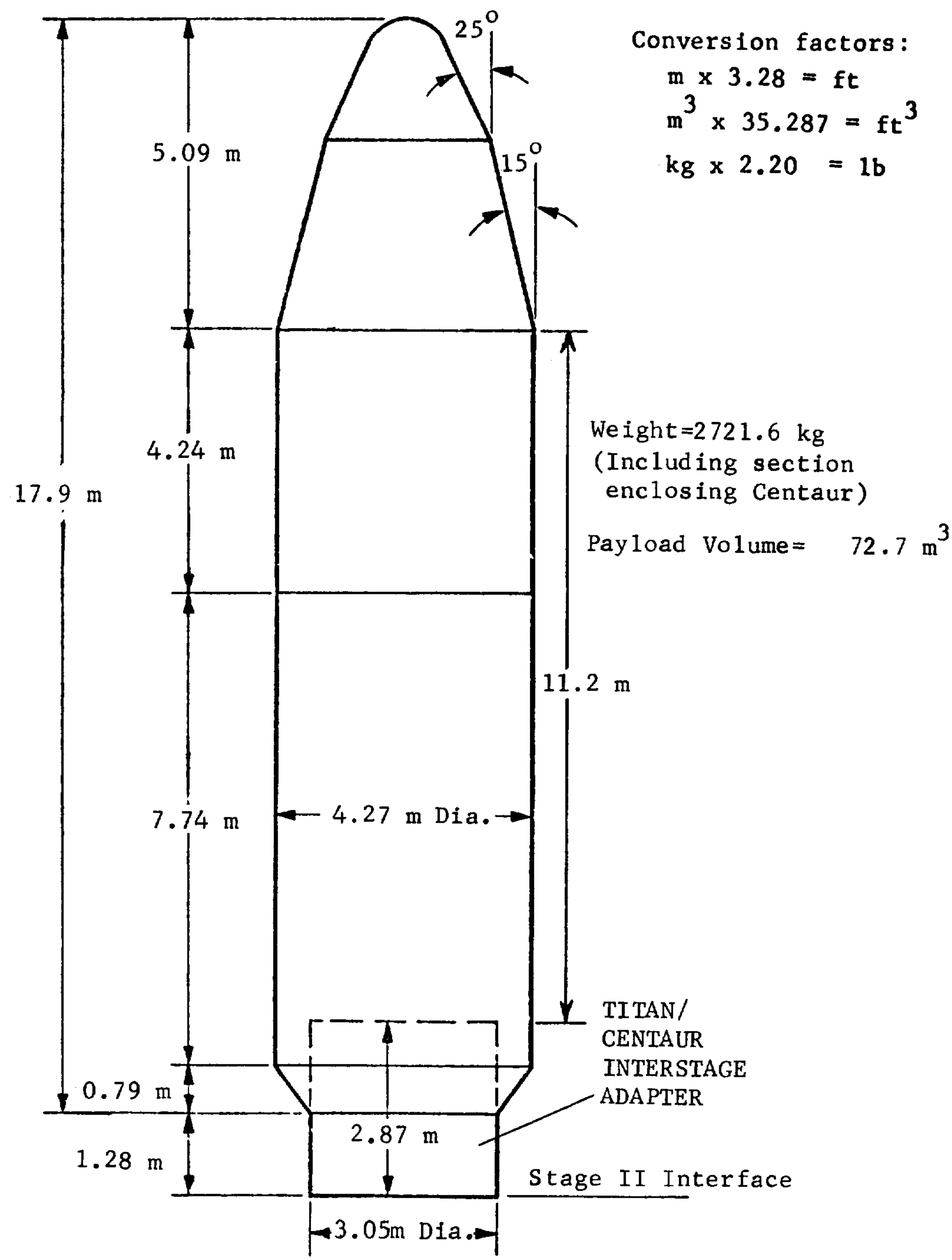


FIGURE 12-15. TITAN/CENTAUR STANDARD SHROUD OVERALL DIMENSIONS

APPENDIX A: DEFINITION OF TERMS

This appendix presents brief definitions of some of the more specialized terms used in this document. The terms are arranged in alphabetical order. Figure A-1 shows terms frequently used in orbital mechanics.

a.u., astronomical unit: The mean distance between Earth and the Sun used as a unit for expressing solar system distances.

Argument of Perigee (Periapsis), The angle, measured in the orbital plane, between the line from the center of attraction to the periapsis and the line from the center of attraction to the point at which the orbit intersects the reference plane when the satellite travels upward (ascending node).

Apsis: The point on an orbit where the distance from the attracting body is either greatest or least. The greatest distance is the apoapsis, while the least is the periapsis.

Aphelion: The apoapsis of an orbit about the Sun.

Apogee: The apoapsis of an orbit about the Earth.

C_3: A measure of total energy (i.e., twice the total energy/unit mass), relative to Earth, remaining after Earth escape, given in km^2/sec^2. The square of the hyperbolic excess velocity.

Declination angle. The angle between a vector and the equatorial plane. The declination angle, as used in Chapter 2, refers to the declination of the [redacted] hyperbolic-excess-velocity vector with respect to the Earth's equatorial plane.

Direction of the Vernal Equinox: A fixed direction in space. The direction of the Sun-Earth position vector when this vector lies in the Earth's equatorial plane and points toward the constellation Aries.

Eccentricity: An indicator of the shape of an orbit. For elliptical orbits, the eccentricity is the difference between apoapsis and periapsis, divided by their sum. For a circular orbit, the eccentricity is zero. For elliptical orbits, the eccentricity approaches unity as the orbit becomes more elongated. For hyperbolic orbits, the eccentricity is greater than unity.

Hyperbolic excess velocity: In the preliminary analysis of interplanetary trajectories, both the Earth-centered escape path and the target planet-centered approach path are hyperbolic. The hyperbolic excess velocity is the velocity at an infinite distance along the asymptote of the hyperbolic path. For Earth escape paths, the hyperbolic excess velocity is approximately the velocity, relative to Earth, as the spacecraft departs Earth's sphere of influence. Conversely, as the spacecraft approaches a target planet, the hyperbolic excess velocity of the approach hyperbola is approximately the velocity relative to the target planet, as the spacecraft enters the sphere of influence.

Inclination: The angle at which the orbital plane intersects the reference plane (usually, the equatorial or ecliptic plane).

Line of nodes: The line formed by the intersection of the orbital plane with the reference plane. See Figure A-1.

Payload: Payload is considered to include all elements normally associated with the spacecraft that must be accelerated to a required final velocity.

Perigee: The periapsis of an orbit about the Earth.

Perihelion: The periapsis of an orbit about the Sun.

Perijove: The periapsis of an orbit about Jupiter.

I_{sp}, specific impulse: Specific Impulse is defined as thrust per mass flow rate. Thus, in SI units, specific impulse has the units newtons/(kg/sec) or equivalently m/sec. In traditional engineering practice it is customary to define specific impulse as thrust per unit weight flow rate, which results in specific impulse units expressed in seconds. The relationship between the two definitions is:

$$I_{sp}\text{ (SI units)} = I_{sp}\text{ (Engr. Units)} \times g$$

where g is the acceleration due to (Earth's) gravity (9.81 m/sec^2).

Sphere of influence: A loosely defined region about a planet within which the planet's gravitational field dominates that of the Sun.

Stable libration points: In a system with two attracting bodies, such as Earth and Moon, rotating about a point, five positions may be found at which a third massless body could be placed in equilibrium under the combined attracting force of the two bodies and the centripetal acceleration associated with the rotation of the system. Only two of these points are stable, such that a small displacement would give rise to forces tending to return the spacecraft to its original position. These stable libration points lie at 60-degree angles on both sides of the line connecting th two attracting bodies, and are equidistant from both. This distance is equal to the distance between centers of the two primary bodies.

Synodic period: The period between two successive conjunctions of two bodies in orbit about a central body.

Trajectories, Type I and Type II: Trajectories for ballistic solar system missions can be divided into two categories characterized by the transfer angles between the Earth at departure and the target at arrival. Type I trajectories are those with transfer angles less than 180 degrees. Type II trajectories are those with transfer angles greater than 180 degrees.

True anomaly: The angle, measured in the orbital plane, between the Periapsis and the current position in the orbit.

V_I, ideal velocity: The ideal velocity is the minimum velocity change needed to perform a mission without velocity losses due to gravity, atmospheric drag and other effects including operational constraints plus the increment necessary to make up these losses. The velocity loss varies with the flight path, the size and configuration of the vehicle and the number and length of vehicle burns. Typical velocity losses for low Earth orbits range from 1.2 to 1.4 km/sec.

V_C, characteristic velocity: As used in this document, characteristic velocity has two compatible meanings, one related to mission requirements and the other related to launch vehicle performance capabilities. First, with respect to mission requirements (e.g., data in Chapters 2 or 3), characteristic velocity is the sum of all velocity increments required to perform the mission starting from a 185 km circular orbit plus the velocity of that orbit, 7797 m/sec, plus any required launch site or launch azimuth penalty from Chapter 4. Second, with respect to launch vehicles (e.g., data in Chapters 5, 7, and 10), the characteristic velocity is the actual total velocity deliverable for each payload using a 185 km circular reference orbit following a due east launch from ETR. These two definitions were chosen to provide a common basis for relating mission requirements and launch vehicle performance data.

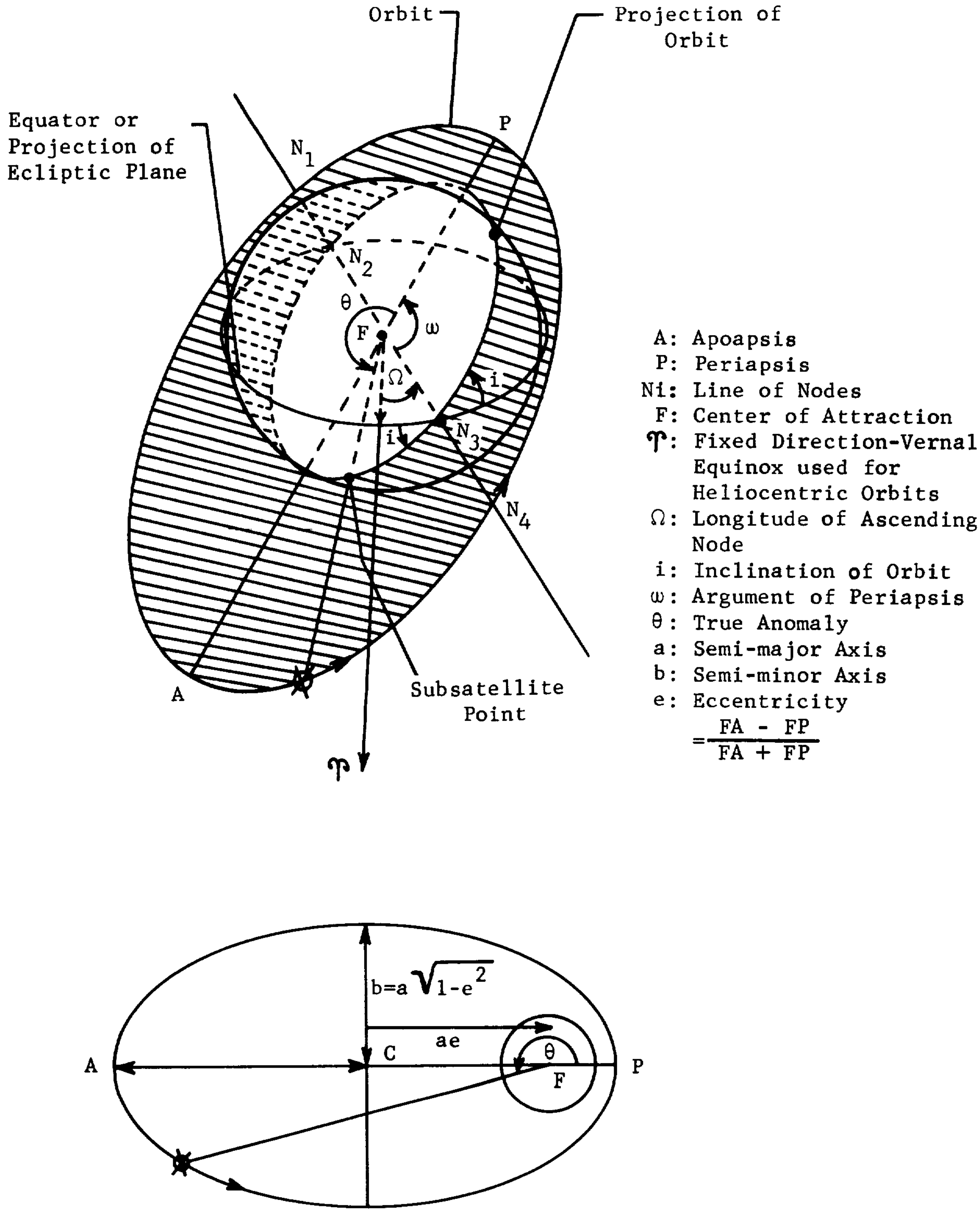


FIGURE A-1. TERMS USED IN ORBITAL MECHANICS

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